JET PROPULSION LABORATORY

CALIFORNIA INSTITUTE OF TECHNOLOGY

PASADENA, CALIFORNIA

INFORMATION ONLY

SURVEYOR LABORATORY
HUGHES AIRCRAFT COMPANY
SPACE SYSTEMS DIVISION

DETAIL SPECIFICATION
SURVEYOR SYSTEM
FUNCTIONAL REQUIREMENTS

RELEASED PRINT			ATTACH.
H REV	3-7-67 REL DATE	[illegible] REL BY	

0791 CC

INFORMATION ONLY

PREPARED [illegible]	DATE 3/3/67	REVISIONS					NUMBER 224510
		LETTER	DATE	AUTH. DOC.	APPROVALS		
[illegible]	3/3/67	S/N H	3-3-7	S/N H RECORD	[illegible]		
APPROVED [illegible]	3/3/67						REVISION H / MODEL
APPROVED							PAGE 1 OF 177 PAGES
APPROVED							CUSTOMER CONCURRENCE

[illegible] 7-64

TABLE OF CONTENTS

A-21 Engineering Test Spacecraft

Page

LIST OF FIGURES

PART I A-21 ENGINEERING TEST SPACECRAFT

1.0 GENERAL

1.1 Scope: This specification, in conjunction with the Surveyor Spacecraft System Design Specification (JPL No. 30240) and the Surveyor Scientific Instrument Interface Specifications, serves to define completely the system functional requirements on the Surveyor spacecraft system including the spacecraft ground equipment, trajectory, prelaunch, launch, and injection requirements. This specification has both the purpose and the authority to define system and subsystem functional performance requirements which, in turn, are to be considered as requirements in the writing of equipment (control item) specifications. Whereas the Design Specification and the Interface Specifications (see paragraphs 2.3.8 through 2.3.14) are contractual documents, this specification is an internal working document. If there is any conflict between this document and the above contractual documents, the most stringent requirements shall apply. All changes, other than deletions, made subsequent to Revision D are indicated by flags in the left margin of this document.

1.2 General Notes

1.2.1 Tolerances: Throughout this specification, tolerances are coded as indicated below. The code designation appears as a superscript following the applicable tolerance.

1) 99 percent confidence level (3σ, RSS of two orthogonal Gaussian variates)

2) 99.8 percent confidence level (3σ, single Gaussian variate)

3) 86 percent confidence level (2σ, RSS of two orthogonal Gaussian variates)

4) 95 percent confidence level (2σ, single Gaussian variate)

5) 40 percent confidence level (1σ, RSS of two orthogonal Gaussian variates)

6) 68 percent confidence level (1σ, single Gaussian variate)

Any tolerance or requirement not so coded shall be assumed as a maximum limit never to be exceeded.

The tolerances specified herein do not include allowances for instrumentation errors, test data reduction errors, and estimation errors due to data sample sizes. Hence these tolerances will not in general be directly applicable for use in test requirements.

1.2.2 * Information Only: Those items marked with an asterisk(*) indicate numbers and data given for information only and not controlled by this document. All other information and data not so marked are considered requirements.

1.2.3 Environment: The performance of the equipment specified in this document shall apply in the environment described in HAC Specification 224800. To the extent that equipment test conditions do not simulate this environment, the performance tolerances allowed herein must be altered to reflect the conditions of test, when deriving test performance tolerances.

1.2.4 Coverage: In conformance with the current revision of the Surveyor System Design Specification, JPL 30240E, this specification is applicable only to the Engineering Test Spacecraft (A-21).

1.3 Definitions: The following definitions are used throughout this specification.

1.3.1 Antenna Axis: Line through the electrical center of the rf beam.

1.3.2 Boresight Axis: The electrical null, peak gain, or rf beam center-line axis of an optical instrument or antenna, defined by hardware markings or optical fixtures.

1.3.3 Mile: Unless otherwise stated, this unit shall be understood as the statute mile (5280 feet).

1.3.4 Miss Vector: The perpendicular distance at the moon between an approach asymptote that would pass through the center of a massless moon and the actual approach asymptote for a massless moon.

1.3.5 Nozzle Centerline: A line passing through the centroids of the nozzle throat and exit areas.

1.3.6 Response Time: Time until response amplitude to a step function input has reached 63 percent of the correct value.

1.3.7 Retro Action Time: The time interval between the condition where the retro thrust has risen to 830 pounds (chamber pressure of 65 psia) on the buildup until the thrust has dropped to 830 pounds (chamber pressure of 65 psia) on the tail-off curve.

1.3.9 Retro Ignition Time: The time interval from the retro ignition signal and the time that the retro thrust level has risen to 5400 pounds (chamber pressure of 300 psia).

1.3.10 Retro Interval: The time interval between the retro ignition signal and the separation of the retro from the spacecraft (approximately twelve seconds after the end of the retro operate interval).

1.3.11 Retro Inert Weight: Weight of the complete retro engine assembly less weight of retro propellant.

1.3.12 Retro Operate Interval: The time interval from the ignition signal to the retro tail-off condition at which the retro thrust has decayed to 3500 pounds.

1.3.13 Retro Thrust-Time Profile: Retro thrust as a function of time over the retro interval.

1.3.14 Retro Useful Specific Impulse: Useful total impulse divided by the propellant burned during the retro interval.

1.3.15 Retro Useful Total Impulse: The integral of thrust as a function of time over the retro interval.

1.3.16 Sensor Group Coordinate System: An orthogonal coordinate system whose axes are defined by optical tooling targets on the flight control sensor group support structure. When attached to the S/C, the sensor group roll, pitch, and yaw axes are aligned to the S/C roll, pitch, and yaw axes, respectively.

1.3.17 Spacecraft Coordinate System: The spacecraft coordinate system is an orthogonal, right-hand Cartesian coordinate system. The three axes, $X_{S/C}$, $Y_{S/C}$, and $Z_{S/C}$ (pitch, yaw and roll, respectively) are defined in paragraphs 1.3.19, 1.3.20, and 1.3.21. The XY-plane is defined as the plane containing three points which are formed by the intersection of the centerlines of the cylindrical optical tooling holes in the column base fittings with the corresponding column base surfaces (points 1, 2, 3 with respect to spacecraft legs 1, 2, 3.) The origin of the coordinate system is a point in the XY-plane which is established by the intersection of the altitudes of the triangle formed by points 1, 2, and 3. Right hand rotation away from the origin is defined as positive rotation.

1.3.18 Spacecraft Dry Landed Weight: Weight of S/C minus the Altitude Marking Radar, the Main Retro Engine, all Vernier Engine fuel Nitrogen gas, and Helium gas.

1.3.19 **Spacecraft Roll (+Z) Axis:** The $Z_{S/C}$ axis is defined as a line which is perpendicular to the XY-plane and which passes through the origin defined in 1.3.17. The positive $Z_{S/C}$ direction is in the direction of the main retro exhaust.

1.3.20 Spacecraft Yaw (+Y) Axis: The $Y_{S/C}$ axis is defined as a line containing the origin and intersection 1. The positive direction is from the origin toward leg No. 1.

1.3.21 Spacecraft Pitch (+X) Axis: The $X_{S/C}$ axis is defined as a line in the XY-plane perpendicular to the $Y_{S/C}$ axis, passing through the origin of the coordinate system with the positive direction such as to form a right-handed coordinate system with the $Y_{S/C}$ and $Z_{S/C}$ axis.

1.3.22 Thrust Vector: The resultant thrust in magnitude and direction produced by the engine.

1.3.23 Vis-Viva Energy (C_3): Twice the total energy per unit mass given by the equation

$$C_3 = V^2 - \frac{2GM}{R}$$

where

R = magnitude of position vector from the earth's center to the S/C at time of injection, km

V = magnitude of the inertial speed at injection, km/sec

$$GM = 3.986032 \times 10^5 \frac{km^3}{sec^2}$$

C_3 is commonly expressed in $\frac{km^2}{sec^2}$

1.3.24 Launch Azimuth (σ_L): The angle, measured at the launch site, from the true north direction to the trajectory plane.

1.4 Abbreviations: The following special or somewhat non-standard abbreviations are used in this specification:

- AFETR - Air Force Eastern Test Range
- CDC - Command and Data Handling Console
- DSIF - Deep Space Instrumentation Facility
- DSS - Deep Space Station (i.e., one of the DSIF stations)
- ETS - Engineering Test Spacecraft
- g - Gravitational acceleration at the surface of the earth (32.174 ft/sec^2) as defined by the U. S. National Bureau of Standards
- HAC - Hughes Aircraft Company
- JPL - Jet Propulsion Laboratory
- OCR - Optimum Charge Regulator
- RSS - Root Sum Squares
- S/C - Spacecraft
- SFOF - Space Flight Operations Facility
- SMS - Scientific Mission Spacecraft
- STEA - System Test Equipment Assembly

2.0 APPLICABLE DOCUMENTS

The following documents are applicable to this specification to the extent indicated herein.

2.1 JPL/NASA Documents

2.1.1 JPL 30240 Surveyor System Design Specification

2.1.2 JPL PD-1 Surveyor/Spacecraft/Launch Vehicle Interface Requirements

2.2 Military Documents: None

2.3 HAC Documents

2.3.1 Drawing 238608; Rocket Engine, Main Retro.

2.3.2 Specification 224600; Detail Specification, Spacecraft Reliability, System, Equipment Group and Subassemblies.

2.3.3 Specification 224601; Reliability Assurance Program for Developmental and Limited Production Equipment.

2.3.4 Specification 224602; Reliability Assurance Program, Surveyor Scientific Payload.

2.3.5 Specification 239505; Phasing and Polarity.

2.3.6 Specification 224800; Environmental Conditions.

2.3.7 Surveyor Spacecraft Monthly Performance Assessment Report, HAC SSD number changes at each issue.

2.3.9 Specification 239303; Alpha Scattering Experiment, Surveyor Scientific Instruments/Spacecraft Interface Specifications.

2.3.10 Specification 239308; Survey Television, Surveyor Scientific Instruments/Spacecraft Interface Specifications.

2.4 Hughes Engineering Orders

The following Engineering Orders (EO's) written against Revision F are reflected in this specification:

51499
51496
51492
5149[illegible]
51488
31684
51580
32202
32211
32215
32216
32217
32218
32219
51577
22[illegible]

3.0 MODE REQUIREMENTS

3.1 Pre-launch

3.1.1 Adjustments: Certain adjustments must be made on the spacecraft prior to launch which are a function of launch conditions (lunar declination, orbital phase, landing location, etc.). Such adjustments should be held to a minimum. In no event should such an adjustment have to be remade during an 8 consecutive day launch period.

Required adjustments are specified for each spacecraft in the corresponding document of the series Numbered 224562 through 224567, inclusive; Launch Parameter Specifications for SC-2 through 7, respectively.

3.1.2 Life on Pad: The auxiliary battery stand time from activation to beginning of pad life shall not exceed 9 days. Pad life commences upon installation of S/C on Centaur and its enclosure by the shroud. After once being installed on the Centaur and enclosed by the shroud, the spacecraft shall not require any additional handling for 10 consecutive days (including 8 possible launch days).

3.1.3 Measurements and Data Transmission: Just prior to launch, the following spacecraft parameters shall be measured and sent to the SFOF at JPL for use by the first DSS in acquiring the spacecraft:

3.1.3.1 Transmitter frequency of both transmitters in both narrow-band VCXO and transponder modes.

3.1.3.2 Transmitter high power output of both transmitters.

3.1.3.3 Receiver frequency of each receiver.

3.1.3.4 Temperature of the operating transmitter.

3.1.3.5 The accuracy of the frequency measurements shall be ±100 cps and the accuracy of the temperature determination shall be ±10°F.

3.1.4 Thermal Capability on Pad: For safety considerations, the spacecraft shall be capable of remaining on stand, under shroud, and without air conditioning for a period of 6 hours, assuming no electrical power is dissipated on the S/C.

3.2 Launch to Injection

This mode is under guidance control of the launch vehicle. It includes the powered ascent and shroud ejection for direct ascent trajectories and powered ascent, shroud ejection, parking orbit coast, and final Centaur ignition for parking orbit trajectories.

3.2.1 Injection Accuracy: The injection errors shall not exceed the capability specified in 3.6.2 and 3.6.3 if the performance specified herein is to be achieved.

3.2.2 Nominal Injection Conditions: The actual injection conditions will be different for each launch day and will be given in the standard trajectories. The nominal injection conditions are as given below.*

3.2.2.1 For direct ascent injections:

Altitude : 90 N.Mi. (167 Km) to 150 n.mi. (278 Km)

Velocity : 35,650 ft/sec (10,870 m/sec) to 36,090 ft/sec (11,000 m/sec)

Elevation Angle (measured from the perpendicular to the earth radius vector) : from -4.0° to 7.5°

Azimuth Angle (measured from powered flight trajectory plane) : 0°

Earth Position : along the line shown in Figure 3-1

3.2.2.2 For parking orbit injections:

Altitude : 90 n.mi (167 Km)

Velocity : 35,940 ft/sec (10,950 m/sec) to 36,090 ft/sec (11,000 m/sec)

Elevation Angle : 2.2°

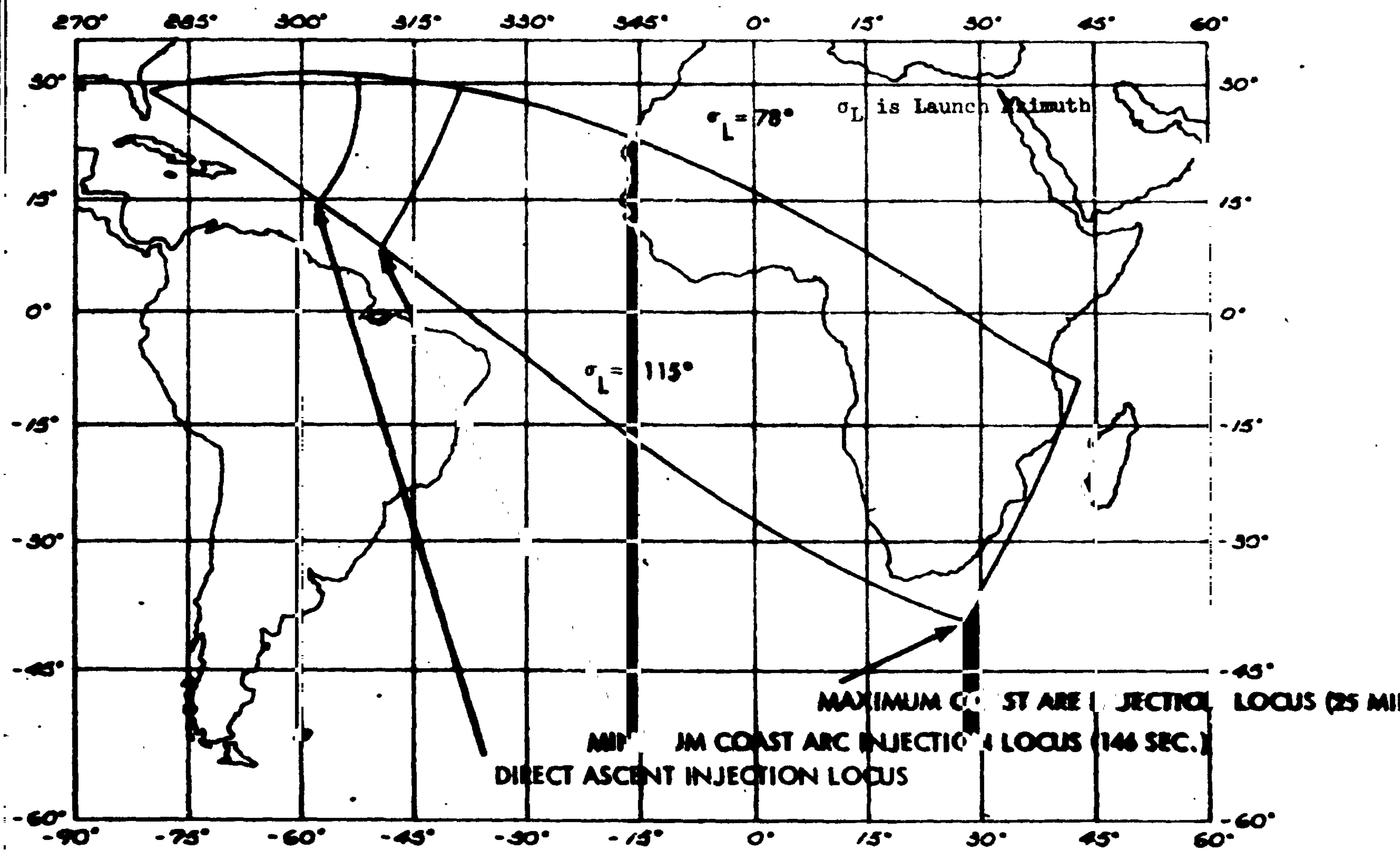


FIGURE 3-1. LOCUS OF INJECTION POINTS

Azimuth Angle (measured from coast orbit plane) : 0°

Earth Position : within the boundary shown in Figure 3-1

3.2.3 Shroud Ejection: The shroud or nose fairing protecting the S/C during passage through the earth's atmosphere shall be ejected prior to injection into the lunar orbit, but not until after the value of ρV^3 is less than 1.9×10^4 lb/sec^3, where ρ is the free stream density (lb/ft^3) and V is the vehicle airspeed (ft/sec). This will nominally occur at a time from launch of 200 seconds*, when the altitude is approximately 360,000 ft. (110 Km) and the velocity relative to the atmosphere is approximately 9,700 ft/sec (2,960 meters/sec).*

3.2.4 Maximum Aerodynamic Heating Parameter Profile

3.2.4.1 Direct Ascent Trajectories

The integrated value of the aerodynamic heating parameter shall not exceed the integrated value on the trajectory shown in Figure 3-2, which is 16,600 $\frac{\text{lb-min}}{\text{sec}^3}$, on any direct ascent trajectory, the maximum instantaneous value of the aerodynamic heating parameter shall not exceed 5020 $\frac{\text{lb}}{\text{sec}^3}$ at any time greater than one minute after shroud ejection.

Additionally, the instantaneous value of the aerodynamic heating parameter may not exceed 500 $\frac{\text{lb}}{\text{sec}^3}$ at any time during the period from one minute until five minutes after shroud ejection, unless the total trajectory integrated value of ρV^3 is less than 16,600 lb-min/sec^3. For trajectories where the integrated ρV^3 is less than 16,600 $\frac{\text{lb-min}}{\text{sec}^3}$, the length of the period during which the limit of 500 $\frac{\text{lb}}{\text{sec}^3}$ on ρV^3 is imposed shall at least be in the same ratio to four minutes as the ratio of the actual integrated ρV^3 is to 16,600. For example, if an actual trajectory results in an integrated value of 10,000 * $\frac{\text{lb-min}}{\text{sec}^3}$, then the limit of 500 $\frac{\text{lb}}{\text{sec}^3}$ may be exceeded beginning at the time of $1' + \frac{10,000}{16,600} \times 4 = 3.41$* minutes after shroud ejection.

3.2.4.2 Parking Orbit Trajectories: During the period beginning one minute after shroud ejection and ending at the time of Centaur main engine second start (MES2), the instantaneous value of the aerodynamic heating parameter. ρV^3, may not exceed 1825 lb/sec^3. Throughout the period from MES2 until the end of significant aerodynamic heating effects, the integrated value of the aerodynamic heating parameter may not exceed 7200 $\frac{\text{lb-min}}{\text{sec}^3}$ and the instantaneous value may not exceed 3700 $\frac{\text{lb}}{\text{sec}^3}$.

3.2.5 Nominal Time from Lift-off to Injection:*

Direct ascent : 11.4 minutes

Parking orbit : 13.3 to 36.4 minutes

3.2.6 Telemetry and Data Transmission:

3.2.6.1* The S/C transmitter frequency shall be monitored as far downrange as possible within limitations of the STEA installed at the launch location; and the frequency-time history shall be transmitted to the SFOF as soon as possible, but with a time lag no greater than 5 minutes (desired).

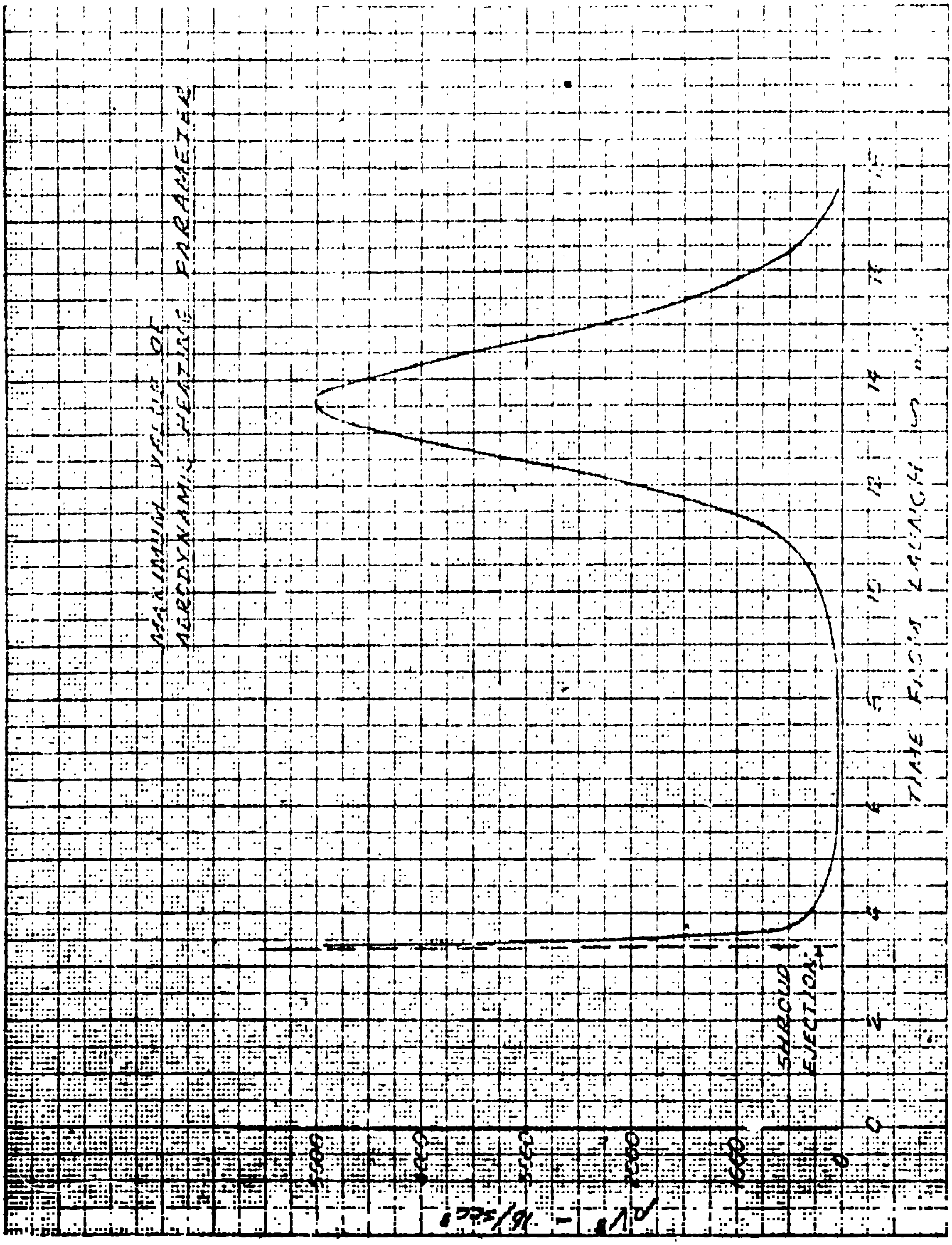


FIGURE 3-2
MAXIMUM VALUE OF AERODYNAMIC HEATING PARAMETER

TITLE	NUMBER 224510	
SURVEYOR LABORATORY HUGHES AIRCRAFT COMPANY SPACE SYSTEMS DIVISION	REVISION H	MODEL S/C
	PAGE 19 OF	PAGES

3.2.6.2* During this mode (launch to injection), the Centaur telemetry channels that are being used by the S/C will be carrying one commutator mode of PCM data as well as shock and vibration data. The shock and vibration data is desired at the SFOF during launch but will be used primarily for post-flight performance evaluation.

3.2.6.3 The S/C transmitter shall be on in its low power mode and shall be modulated with certain S/C engineering data with a carrier modulation index not to exceed 0.4 rad.

3.2.7 Angular Rates: The maximum angular rates throughout this phase shall be less than 5 deg/sec(1), except for specific attitude maneuvers wherein rates up to 12 deg/sec may exist for periods up to one second.

3.3 Separation and DSIF Acquisition

3.3.1 Pre-separation: At Centaur burnout, several spacecraft operations are initiated by command signals from the Centaur.

3.3.1.1 Pre-separation S/C Operations

3.3.1.1.1 Extension of landing gear

3.3.1.1.2 Extension of omnidirectional antennas

3.3.1.1.3 Transmitter high power on

3.3.1.1.4 N_2 gas jet valve amplifiers enable (initiated by the separation signal).

3.3.1.2 In addition, the telemetering channel in the Centaur which is to be utilized for spacecraft PCM data shall be instrumented to provide the following information which shall be sent to the SFOF to be relayed to the DSIF stations:

Omnidirectional antenna extend verification signal

Leg extend verification signal

Transmitter power

3.3.1.3 Throughout the above S/C pre-separation operation, the angular rates shall be less than 3 deg/sec and the angular acceleration shall be less than 1 deg/sec^2.

3.3.2 Separation: The actual separation shall be verified by a signal in the Centaur which can be sent from AFETR to the SFOF.

3.3.2.1 Angular rates at separation: The Centaur vehicle, plus all Centaur/spacecraft separation mechanisms, shall not impart a rotational velocity greater than 3.0 deg/sec(1) to the spacecraft at separation.

3.3.2.2 Uncertainty in separation velocity imparted to spacecraft shall be included in injection errors. See 3.2.1.

3.3.2.3 The actual separation shall be instrumented via the Centaur telemetry in such a manner that the relative separation rates (both angular and rectilinear) can be determined in post-flight evaluation to within an accuracy of ± 10%(2).

3.3.2.4 Longitudinal separation velocity: The Centaur vehicle and separation mechanism shall not impart a relative velocity at separation to the spacecraft of less than 0.4 ft/sec.

3.3.2.5 Separation Shock: The separation shall not impart an acceleration to the spacecraft of more than 2 g's(2).

3.3.3 Post-separation: The flight control system is in the rate stabilization mode, operating with the cold gas jets. Separation angular rates should be reduced and maintained near zero.

3.3.3.1 Attitude accuracy: Not critical

3.3.3.2 Automatic solar panel deployment: Automatic solar panel deployment shall be initiated when the spacecraft separates from the Centaur. Automatic operations shall include: (1) unlocking the solar panel and roll axis so that they are free to move from their launch positions; (2) stepping of the solar panel axis 85 degrees so that the solar panel is positioned perpendicular to the spacecraft Z axis; and (3) stepping the roll axis 60 degrees counterclockwise after deployment is complete so that the roll axis is in its proper transit position. The complete deployment sequence shall nominally require no more than 9 minutes and 40 seconds for completion.

Automatic sun acquisition shall be initiated by the spacecraft at 51 ± 1 seconds after separation.

3.3.3.3 Spacecraft angular rates: The flight control system shall be capable of reducing spacecraft angular rates to approximately 0.1 deg/sec within 50 seconds after separation.

3.3.3.4 Centaur - Spacecraft separation distance: The post separation Centaur maneuvers shall be such that the minimum separation distance at 5 hours after injection shall be greater than 336 Km.

3.3.4 DSIF Acquisition: Acquisition of the spacecraft by the DSIF takes place as follows:

3.3.4.1 The DSS antenna is pointed in the expected direction of the spacecraft.

3.3.4.2 The DSS receiver is tuned to the nominal frequency plus doppler shift of the spacecraft transmitter signal.

3.3.4.3 When a signal is received, the DSS receiver locks on in phase, and the DSS antenna is switched to angle track.

3.3.4.4 The DSS transmitter is then turned on and slewed in frequency. When the received frequency passes through the operating range of the spacecraft receiver phase lock loop, it locks on and the spacecraft receiver and transmitter track the DSS transmitter frequency. The accomplishment of phase lock can be noted by a sudden shift in frequency of the spacecraft transmitter. This may cause the DSS receiver to lose the signal from the spacecraft. If so, the DSS receiver must be relocked.

3.3.4.5 The DSS transmitter is now tuned to the center of the operating range of the spacecraft transponder phase locked loop by nulling the static phase error.

3.3.5 Information Flow Necessary for Acquisition:

3.3.5.1 Tracking data from AFETR downrange stations and Centaur determined injection parameters shall be sent to JPL for use in trajectory determination.

3.3.5.2 In addition, the following spacecraft performance data obtained from AFETR prelaunch measurements and/or Centaur telemetry shall be TWX'd to JPL:

3.3.5.2.1 Transmitter frequencies (for narrow-band VCXO and transponder modes)

3.3.5.2.2 Transmitter power output.

3.3.5.2.3 Transmitter temperature.

3.3.5.2.4 Receiver frequencies.

3.3.5.2.5 Leg extension confirmation.

3.3.5.2.6 Omni-antenna extension confirmation.

3.3.5.2.7 Separation confirmation.

3.3.5.3 The following data shall then be sent from JPL to the appropriate DSIF station:

3.3.5.3.1 DSS antenna pointing data in terms of either hour angle, declination angle and time, or azimuth angle, elevation angle, and time.

3.3.5.3.2 Doppler shift versus time.

3.3.5.3.3 S/C transmitter frequencies.

3.3.5.3.4 S/C receiver frequencies.

3.3.5.3.5 S/C transmitter power.

3.3.5.3.6 Confirmation of omni-antenna extension, leg extension and separation (if available in time).

3.3.6 Timing:

3.3.6.1 Time required for all pre-separation spacecraft operations: less than 22 seconds.*

3.3.6.2 Time between injection and first possible DSIF acquisition: varies between 0 and approximately 20 minutes for parking orbit injections and between 10 and 14 minutes for direct ascent injections.

3.3.6.3 Time required for first DSIF acquisition: less than 5 minutes.

3.3.7 Post Acquisition Sequence

The post acquisition sequence shall be constrained so that the total continuous unmonitored "on time" for the high power transmitter shall not exceed 60 minutes.

3.4 Attitude Reference Acquisition

As a part of the automatic sun acquisition sequence, the vehicle shall be commanded to acquire, track and control the vehicle attitude to the sun in two axes (pitch and yaw). During a subsequent star acquisition procedure, the spacecraft is commanded to acquire, track, and control the vehicle attitude to a star (Canopus) in the third axis (roll). Sun lock-on is required to permit continuous battery charging via the solar panel throughout transit. Sun and star lock-on are required to provide a known and accurate vehicle attitude prior to midcourse and terminal descent maneuvers, and to maintain a fixed vehicle attitude relative to the sun for thermal control purposes.

3.4.1 Sun Acquisition:

3.4.1.1 Sequence

3.4.1.1.1 Fifty-one (51) seconds after separation, sun acquisition is initiated by a command from the flight control programmer, which causes a vehicle roll maneuver of -0.5 deg/sec* and continues until the sun comes into the field-of-view of the secondary sun sensor which is affixed approximately on the spacecraft roll-pitch plane.

3.4.1.1.2 When this occurs, the roll command is removed and a yaw command is initiated to bring the primary sun sensor line of sight toward the sun.

3.4.1.1.3 When the sun falls into the field-of-view of the primary sun sensor, a lock-on signal is generated. This signal switches vehicle attitude control to the primary sun sensor and also serves to indicate (via telemetry) the completion of sun acquisition.

3.4.1.2 Loss of lock: Once sun lock has been established, loss of this sun lock-on signal shall automatically switch the attitude control back to an inertial attitude hold mode.

3.4.1.3 Attitude control accuracy: As specified in 4.3.1.1

3.4.1.4 Timing

3.4.1.4.1 Maximum time to acquire sun: 18 minutes.

3.4.2 Star (Canopus) Acquisition: The star acquisition mode is command initiated after completion of the sun acquisition and at least 2 hours prior to the midcourse correction.

3.4.2.1 Sequence

3.4.2.1.1 The star acquisition command starts a vehicle positive roll (clockwise looking at the S/C from the direction of the sun) of 0.5 deg/sec* until a star of the correct brightness falls into the sensor field-of-view.

3.4.2.1.2 When this occurs, a lock-on signal is generated which stops the 0.5 deg/sec roll rate and switches the vehicle roll control to the star sensor error signal.

3.4.2.2 Loss of lock: Once star lock has been established, loss of this lock-on signal shall automatically switch the vehicle roll control to an inertial attitude hold mode.

3.4.2.3 Canopus verification: To verify that the correct star has been acquired, the S/C is commanded to roll up to 720° in one continuous roll. During this roll, the unthresholded star intensity signal, as well as the normal thresholded signal, is monitored. From these signals, a star map is made and Canopus identified. The capability for performing at least 4 of these verifications shall be provided. This verification shall be performed before the normal star acquisition mode is initiated.

3.4.2.4 Attitude control accuracy: As specified in 4.3.1.2.

3.4.2.5 Acquisition time: 38 minutes maximum (assuming a normal verification and excluding the time for studying the star map on the ground).

3.5 Coast Mode I

This mode commences on completion of the sun/star acquisition and terminates just prior to the midcourse correction. During this mode, the S/C coasts with its attitude servoed to the sun and star with the low power transmitter on in the transponder mode to permit continuous two-way doppler tracking and to provide continuous low bit rate telemetry. Periodic samplings of certain engineering data (not to exceed 10 during the transit portion of the mission) shall be made, some of which will require switching on the high power transmitter at those times.

3.5.1 Attitude Accuracy: While in the coast mode the allowable error is ±2°. One-half hour prior to the midcourse maneuver, however, the attitude requirements are as specified in 4.3.1.1 and 4.3.1.2 of this document.

3.5.2 Doppler Tracking: Throughout this mode, the S/C angular position and range rate is being measured by one of the DSIF stations. The range rate is a two-way doppler measurement requiring a phase locked transponder in the S/C. It is required that this be accomplished on the omni-antenna with the low power transmitter.

3.5.2.1 Operating range: This two-way doppler link must operate over ranges from 90 n.mi. (167 Km) to 219,500 n.mi (406,700 Km) (including Coast Mode II) and relative velocities between plus and minus 10 KM/sec (32,800 ft/sec).

3.6 Midcourse Correction

This mode involves an attitude maneuver to align the vehicle roll axis in the computed direction for the velocity correction, an accurately controlled velocity correction and a reverse attitude maneuver to reacquire the sun and star. The S/C mechanization, with the possible exception of the vernier fuel loading, shall be capable of two such midcourse corrections. However, the time between a subsequent ignition of the vernier engines and the termination of the last previous ignition of the engines shall be at least one hour.

3.6.1 Correction Computation: From the S/C tracking data, its orbit and predicted lunar miss shall be computed by the SFOF Computing Facility. See Section 6.4 of this specification for details.

3.6.2 Correction Capability: Capability for an automatically timed velocity correction of up to 50 meters/sec (164 ft/sec) shall be provided. This provides a nominal capability of correcting a miss at the moon of 9610 Km (5170 n.mi.) perpendicular to the approach trajectory. The emergency capability for velocity corrections greater than 51 meters/sec, but with reduced accuracy, shall be provided (with the exception of vernier fuel) through the use of manual burning time control from the ground. The minimum possible velocity correction command shall be 2.0 meters/sec (6.6 ft/sec). The spacecraft design shall be such as to accommodate midcourse maneuvers between zero and 50 meters/sec.

3.6.3 Correction accuracy: For all automatically timed corrections within the capability of paragraph 3.6.2, the residual miss on the lunar surface due to S/C mechanization shall not exceed 100 Km (1) (53.9 n.mi.) for a vertical approach trajectory or 160 Km (1) (86.25 n.mi) for a 45 degree approach trajectory. These requirements shall apply only to a nominal 66 hour trajectory wherein the midcourse correction is made no sooner than 20 hours after injection. In order to meet these requirements, the following accuracies must be met:

3.6.3.1 The actual thrust vector must be controlled such that the resultant velocity increment is within ±0.7 degrees(1) of the computed direction.

3.6.3.2 For incremental velocity changes up to 100 ft/sec (30.5 m/sec), the error in the magnitude of the velocity change shall be less than ±1.3 ft/sec(2) and for larger velocity increments, the error shall scale linearly.

3.6.3.3 Acceleration level: as specified in 4.4.1.

3.6.4 Timing (Midcourse Correction):

3.6.4.1 Time of maneuver: As early as 4 hours and as late as 40 hours after injection. The nominal correction time shall be 15 hours after injection for direct ascent and 20 hours for parking orbit. However, all correction accuracy computations shall be based on a nominal 20 hour correction.

3.6.4.2 Initiation of attitude orientation: <15 min prior to velocity change.

3.6.4.3 Time required for attitude orientation: 12 min max.

3.6.4.4 Max. time available for velocity change: 51.2 sec. in automatically timed mode.

3.6.4.5 Time required for normal reacquisition of sun and star: 15 min max (excluding any Canopus verification time).*

3.6.4.6 Required lead time allowance between the end of the last midcourse guidance computation and the performance of the midcourse correction: at least 2.0 hours.

3.6.4.7 Allowable time between actual velocity change and precomputed time for the velocity change: At least ±5 minutes. This is a variable to be computed by the m/c correction computer program.

3.6.4.8 Timing Accuracy: The error between the command time interval and the actual time interval shall not exceed the following:

a. 0.2 seconds plus 0.02% of the command interval magnitude (1) for attitude orientation maneuvers.

b. 0.025 seconds plus 0.02% of the command interval (1) for velocity increment duration.

3.7 Coast Mode II

This mode is essentially the same as Coast Mode I except it occurs at greater S/C to earth ranges. This additional tracking data is required to reestablish the orbit after the midcourse correction and to accurately determine the landing location. This post-midcourse tracking should be continued until just prior to the pre-retro maneuver.

3.8 Pre-retro Maneuver

This mode includes: (a) the deployment of the planar array, (b) spacecraft maneuvers to align the main retro thrust axis with the velocity vector and the high gain planar array with the spacecraft-earth line to transmit television pictures during lunar approach and, (c) the subsequent coast period prior to retro-ignition during which descent TV pictures are transmitted to the earth.

3.8.1 Required Action: The following items shall be accomplished during this mode:

3.8.1.1 Command all TV temperature controls on approximately 3 to 8 hours prior to computed ignition time, depending on S/C supply voltage.

3.8.1.2 Command deployment of the planar array to its landing position if less than 45° or to 45° if the required position is greater than 45°. This shall be done while still locked onto the sun and star.

3.8.1.3 Command computed roll maneuver as first step in bringing thrust axis in line with velocity vector. Pitch and yaw attitude controls remain on sun sensor hold. A less desirable alternative is to do pitch or yaw maneuver first.

3.8.1.4 Command vehicle pitch or yaw maneuver to bring the thrust vector in line with the velocity vector.

3.8.1.5 Command second roll maneuver to align the planar array with earth.

3.8.1.7 Command vernier engine nominal thrust bias (if necessary).

3.8.1.8 Command setting of time delay between altitude marking signal and engine ignition.

3.8.1.9 Command altitude marking radar on.

3.8.2 __Timing__: (Pre-retro Maneuver)

3.8.2.1 Time required to erect the planar array: 10 minutes max.

3.8.2.2 Initiation of maneuver: No more than 33 minutes prior to computed retro ignition time.

3.8.2.3 Time for complete normal attitude maneuvers: 17 minutes max (incl. 60 sec time between successive maneuvers).

3.8.2.4 Additional time required for emergency or non-normal attitude maneuver, if required, will come out of the time normally allotted to the TV.

3.8.2.5 Time by which all normal attitude maneuvers must be completed: 15 minutes prior to computed retro ignition time.

3.8.2.9 Command setting of retro time delay: at least 6 minutes prior to retro ignition.

3.8.2.10 Altitude marking radar power to "on": 5 minutes prior to retro ignition.

3.8.2.11 Marking radar enable: Marking radar is turned to enable at the altitude specified in 4.6.2.4. To accomplish this, the command shall be sent at the computed time ±10 seconds.

3.8.3 Accuracy:

3.8.3.1 The pre-retro maneuver shall be such that the pointing accuracy requirement of 3.9.3.2 can be met during retro burning.

3.8.3.2 Throughout this phase, the angle between the planar array electrical axis and the true S/C - DSIF line shall be less than 2.3 degrees.(2)

3.8.3.3 Timing Accuracy: As given in 3.6.4.8.

3.9 Main Retro Descent

This mode begins with the altitude marking radar ignition signal and ends with ejection of the spent main retro case.

3.9.1 Action Sequence: The following automatic sequence shall be initiated by the marking radar signal:

3.9.1.1 At a nominal slant range of 60 miles the altitude marking radar triggers a timer which in sequence starts the vernier engines, ignites the main retro, and applies power to the radar altimeter and doppler velocity sensor.

3.9.1.2 Vernier engines started by timer after preset delay (3.6.1.8).

3.9.1.3 Main retro ignited approximately 1 sec following vernier ignition.

3.9.1.4 RADVS turned on at 0.55 sec following retro ignition.

3.9.1.5 Main retro burnout is signalled by the acceleration falling to 3.5 ± 0.5g. This signal initiates the following: (a) vernier thrust level increase to maximum programmed level, (b) starts a timer to blow retro separation nuts and to permit pitch and yaw attitude control to be switched from inertial to doppler velocity reference if or when the doppler velocity reliable signal is present. The inertial switch shall not signal retro burnout when the instantaneous thrust-to-mass ratio is greater than 161 fps^2.

3.9.2 Timing: (Main Retro Descent)

3.9.2.1 Main retro ignition time: Approximately 3 minutes before touchdown.

3.9.2.2 Time between altitude marking radar signal and vernier ignition: Radio commandable from 0 to 20 seconds or greater.

3.9.2.3 Time between vernier ignition and main retro ignition: 1.1 sec ± 0.1 sec.(2)

3.9.2.4 Main retro action time: Approximately 42 sec.

3.9.2.5 Time from inertia switch signal to main retro ejection signal: 12 ± 0.1 sec.(2)

3.9.2.6 Time from main retro ejection signal to first possible attitude reference change: 2.15 ± 0.1 sec.(2)

3.9.3 Accuracy

3.9.3.1 Accuracy on time delay between altitude marking signal and actual main retro ignition (as defined in 1.3): 0.055 sec required.(2)

3.9.3.2 At the time of main retro ignition and throughout main retro burning, the actual main retro thrust axis shall be aligned to the actual initial velocity vector plus offset (if applicable) within ±1 degree.(1)

3.9.3.3 Attitude rates during retro burning: As specified in 4.3.2 of this specification.

3.9.3.4 Throughout main retro burning the angle between the planar array electrical axis and the true S/C DSIF line shall be less than 2.5 degrees(2) with maximum pitch, yaw and roll disturbances.

3.9.4 Burnout Conditions: The nominal S/C conditions at main retro burnout vary from flight to flight as a function of lunar distance, declination, time of flight, landing location, etc. For all these variables, the maximum and minimum nominal burnout conditions for a normal and a 25° approach are given below (assuming no doppler attitude control until after main retro burnout).

3.9.5 Unbraked Impact Approach Angle: The S/C shall be capable of a soft landing for unbraked impact approach angles up to 25 degrees off vertical.

MAIN RETRO BURNOUT CONDITIONS

Item	For Normal Approach				For 25° Approach			
	Maximum		Minimum		Maximum		Minimum	
	Nominal	$\pm3\sigma$	Nominal	$\pm3\sigma$	Nominal	$\pm3\sigma$	Nominal	$\pm3\sigma$
Altitude (Kft)	47	±9	22	±9	47	±9	28	±9
Slant Range (Kft)	49	±9	31	13	61	±12	39	±13
Velocity Magnitude (fps)	525	±125	200	±125	525	±125	250	±125
Angle Between Velocity and Lunar Vertical	0°	±16	0°	±45°	23°	±16	14°	±25°
Angle Between Roll Axis and Lunar Vertical	0°	±1°	0°	±1°	25°	±1°	25°	±1°

3.10 Vernier Descent

The vernier descent phase begins with the ejection of the main retro engine case and ends with the vernier engine shut-off which nominally occurs at an altitude of 14 feet.

3.10.1 Action Sequence: The following automatic sequence controlled by the spacecraft shall be followed in this phase:

3.10.1.1 At a fixed time delay of 2.15 ±0.01 seconds following the retro case ejection signal (to allow time for the case to clear the vehicle), the pitch and yaw attitude control and the vernier engine thrust control is "armed" to permit switching to vernier descent control modes when the radar reliable signals are present.

3.10.1.2 When the doppler velocity reliable signal is present and the attitude control is "armed", the pitch and yaw attitude control shall be switched to the doppler velocity reference and the vehicle attitude is controlled by servoing the components of velocity normal to the vehicle axis (V_x and V_y) toward zero. The S/C shall be designed such that the doppler reference can be used for attitude control prior to retro burn-out if so desired by minor circuit revisions.

3.10.1.3 If the altimeter reliable signal is not present when the thrust control is "armed", the thrust command is switched from maximum programmed thrust (see 4.4.3) to minimum programmed acceleration (see 4.4.4.1) and is held there until the altimeter reliable comes on.

3.10.1.4 When the altimeter reliable signal is present, the thrust control shall be switched from the fixed thrust command to the programmed velocity-slant range trajectory control mode. In this mode, doppler velocity information is used to provide proper error signals to the velocity control system to maintain the desired slant range/velocity ratio prescribed by the programmed descent trajectory.

3.10.1.5 When a velocity (V_z) of 10 ft/sec is reached (nominally at an altitude of 43 ft, the constant velocity control mode is initiated. Attitude control is switched back to the inertial reference and engine thrust is servoed so as to achieve and maintain a 5 ft/sec constant value.

3.10.1.6 When an altitude of 14 feet is reached, the vernier engines shall be shut off, thus terminating this mode.

3.10.2 Timing: (Vernier Descent)

3.10.2.1 Time between initiation of attitude maneuver and alignment of vehicle roll axis to velocity vector: 9 sec max (for 45° maneuver).

3.10.3 Accuracy: In order to meet the requirements of 3.11.2, the following conditions shall be met:

3.10.3.1 Vernier engine cut-off altitude: 14 ft ± 4.5 ft(2) from the lunar surface to the altimeter antenna boresight plane.

3.10.3.2 Vertical velocity at vernier engine cut-off: 5 ft/sec ± 1.5 ft/sec.(2)

3.10.3.3 Altitude at switch-over to inertial attitude control: 43 ± 15 ft.(2)

3.10.3.4 Lateral velocity at vernier engine cut-off: 0 ± 4.0 ft/sec.(1)

3.10.3.5 Angle between roll axis and local vertical at vernier engine cut-off: 0 ± 4.8 deg.(1)

3.10.3.6 Total angular velocity after vernier engine cut-off: <3.2 deg/sec. (1)

3.10.3.7 Roll attitude: The roll attitude throughout this phase shall be as specified in 4.3.1.4 of this specification.

3.11 Touchdown

After vernier engine cut-off, the S/C free falls to the lunar surface.

3.11.1 Timing: The time from vernier engine cut-off until first contact with the lunar surface shall be less than 2 sec.

3.11.2 Pre-Touchdown Conditions

The S/C descent shall be controlled such that the following conditions are in effect at the instant prior to touchdown (contact with lunar surface).

3.11.2.1 Longitudinal velocity: < 15 ft/sec.(2)

Longitudinal refers to the z-axis of the spacecraft coordinate system.

3.11.2.2 Lateral velocity: <5.0 ft/sec.(1)

Lateral velocity is the component of total velocity which lies in the x-y plane of the spacecraft coordinate system.

3.11.2.3 Angle between roll axis and local vertical: <7 deg.(1)

This angle is measured in the plane defined by the S/C velocity and local vertical.

3.11.2.4 Magnitude of total angular velocity: <3.2 deg/sec.(1)

3.11.3 Lunar Surface: For touchdown considerations, the lunar surface shall be as specified in 3.13.3.5 of the Surveyor Design Specification, JPL No. 30240.

3.11.4 Touchdown Capability

The structural mechanization of the S/C shall be such that landings under the following conditions can be sustained without damage to the S/C that would impair the conduct of lunar operations. Note that "horizontal" and "vertical" in this paragraph are not equivalent to "lateral" and "longitudinal" in paragraph 3.11.2.

3.11.4.1 Vertical velocity: ≤ 20 ft/sec

3.11.4.2 Horizontal velocity: ≤ 7 ft/sec

3.11.4.3 Angle, S/C Roll Axis to Vertical: ≤ 8 deg

3.11.4.4 Angular rate, angle of 3.11.4.3: ≤ 4 deg/sec

3.12 Postlanding Engineering Verification: The postlanding engineering verification phase begins with lunar contact when the final spacecraft kinetic energy is absorbed by the landing gear, and is terminated when the solar panel acquisition of the sun (for the case of a lunar day landing), the planar array acquisition of the earth, and the postlanding engineering assessment are completed. Unless indicated otherwise, the requirements specified refer to day or night landings. The timing specified assumes the absence of operational thermal constraints. A mid-day landing may be subject to the constraints specified in 4.10, thereby requiring an extended period for completion.

3.12.1 Operations: After touchdown, but before the preparation for the lunar experiments, the following operations are required to prepare the spacecraft for the lunar phase of operation:

3.12.1.1 Verification of Communication Capability: It is required that the ground receiver maintain phase lock during the touchdown period. If phase lock is lost for any reason, reacquisition will normally be complete within 20 seconds after touchdown.

3.12.1.2 Termination of Transit Operations: Terminal descent phase subsystems or functions not required during the postlanding lunar operations shall be turned off by ground command.

3.12.1.3 Assessment of Basic Bus Touchdown Survival: An assessment of the operating condition of as many basic bus functions as is possible is required for mission evaluation and lunar operational planning.

3.12.1.3.1 Initial Postlanding Engineering Assessment: The initial assessment shall provide preliminary evaluation of the signal processing, narrow band transmission, and the power and thermal status of the spacecraft to the extent possible. The purpose of the assessment is to provide an immediate verification of successful soft-landing as well as the capability of the spacecraft to support subsequent operations.

3.12.1.3.2 Subsequent Engineering Assessments: Additional assessments are required to verify the functional capability of the antenna and solar panel positioners and the solar panel and battery charging circuitry (day landing only) and the planar array.

3.12.1.3.3 Mode Television: The television camera may be operated in the 200 Line Scan mode to provide preliminary data related to the lunar surface and the landed condition of the spacecraft.

3.12.1.4 Orientation of the Solar Panel and Planar Array: The optimum orientation procedure to be used is dependent upon the landing site location and solar lighting conditions.

3.12.1.5 Determination of S/C Landed Attitude (day landing only)

3.12.1.6 Assessment of Basic Bus Capability to Support Lunar Experimentation.

3.12.2 Timing

3.12.2.1 Initiation: Immediately after touchdown.

3.12.2.2 Time for Completion

3.12.2.2.1 The verification of communication and the termination of transit operations will be completed within approximately 2-1/2 minutes. RADVS power shall be turned off within 2 minutes after touchdown. Execution of the thrust phase power off commanded must be verified prior to RADVS power turnoff.

3.12.3 Accuracy

3.12.3.1 Solar Panel and Planar Array Positioning (Day Landings) For the purpose of determining the spacecraft attitude, the following positioning accuracies are required: The initial positioning of the solar panel shall be such that the secondary sun sensor boresight axis will be parallel to the spacecraft-sun line within ±0.75 degree(3). The initial positioning of the planar array shall be such that the antenna electrical axis will be parallel to the spacecraft/DSIF line within ±1.0 degree(3). This implies a received power meter on the ground that will indicate relative power levels to within at least 0.3 db.

3.12.3.2 Post Landing Attitude Determination (Day Landings) The attitude of the S/C in celestial and selenographic coordinates shall be determined from the solar panel and planar array gimbal angles with sufficient accuracy to predict the following:

1) Local vertical to 4 degrees(3) in S/C coordinates for earth-sun angles greater than 15 degrees.

2) Sun angle to 2 degrees(3) is S/C coordinates for earth-sun angles greater than 15 degrees and assuming solar panel and planar array positioning at 24 hour intervals.

3) Variations in local vertical in S/C coordinates in a 24 hour interval to 4 degrees.

4) Time to terminator to ±8 hours for earth-sun angles greater than 15 degrees assuming a known horizon in S/C coordinates with a maximum of 15 degree slope.

3.13 Lunar Experimentation: The lunar experimentation phase begins with the completion of the solar panel and planar array positioning and engineering assessment and continues until the end of the mission. Unless indicated otherwise the requirements specified refer to day landings.

3.13.1 Survey Television

3.13.1.1 Spacecraft Observations: At appropriate times during the lunar mission, television will be used to aid in the general engineering evaluation of the spacecraft status.

3.13.1.2 Payload Observations:

At appropriate times during the lunar mission, television will be used to aid in the evaluation of the Alpha Scattering Instrument. Deployment of the Alpha Scattering Instrument will be aided by television observations.

3.13.2 Engineering Interrogations

3.13.2.1 Sampling of the Mode 4 or Mode 5 Commutator Data: The Mode 4 or Mode 5 commutator data shall be sampled at least once each 1.5 hours (minimum of 30 frames) during the non-critical periods of the lunar phase. It is desirable to transmit data continuously during the following periods:

3.13.2.1.1 Postlanding engineering verification.

3.13.2.1.2 Repositioning of planar array and solar panel.

3.13.2.1.3 Terminator interval.

3.13.2.2 Redundant Data Requirements: Engineering commutator Modes 1, 2, 4 and 5 shall be sampled at least once each 6 hours (minimum of 30 frames each).

3.13.3 Solar Panel and Planar Array Repositioning.

3.13.3.1 Solar Panel Repositioning: The solar panel will normally be repositioned to track the sun a minimum of each earth day during the lunar day.

3.13.3.2 Final Solar Panel Positioning: The solar panel will be repositioned such as to limit the output voltage to a safe level at sunrise.

3.13.3.3 Planar Array Repositioning: The planar array will normally be repositioned a minimum of each earth day during the lunar day to reduce pointing errors resulting from lunar librations.

3.13.3.4 Planar Array Repositioning Accuracy: The repositioning shall be accomplished to provide electrical axis pointing erros consistent with the spacecraft transmission modes in operation.

3.13.3.4.1 The electrical axis pointing error during the transmission of gimbal angle data for attitude determination shall be less than 1.0 degree. The pointing error during television operation shall be less than 1.4 degrees (0.5 db).

3.13.4 Spacecraft Power and Thermal Control Operations

3.13.4.1 Standby Mode: In order to most efficiently utilize the spacecraft compartment A and B thermal capability and to maintain the compartment temperatures below 125°F under adverse equatorial lunar noon environmental conditions, a capability to establish a standby mode from a data transmission mode or a data transmission mode from a standby mode by ground command is required. In the standby mode only the command receivers will be operating with fill-in words interrupted and the electrical power dissipated as heat in compartment A and B shall be limited to approximately 3 watts and 0 watts for day or night landings respectively.

3.13.4.2 Pre-Sunset Compartment Heating: To maximize post-sunset operating capability, solar panel power shall be used to raise compartment tray temperatures to the thermal switch closure temperature.

3.13.4.3 Compartment A and B Heater Control: A capability for 3 states of compartment heater control shall be provided by ground command (day or night).

3.13.4.3.1 Compartment A and/or B heaters automatic.

3.13.4.3.2 Compartment A and/or B heaters on.

3.13.4.3.3 Compartment A and/or B heaters off.

3.13.4.4 Automatic Battery Charging: The power system shall be capable of automatically charging the battery upon solar panel illumination when the initial state of the spacecraft is the standby mode described in paragraph 3.13.4.1 and the battery is less than fully charged (day or night landing).

3.13.4.5 Minimum Battery Charging Time: A minimum float charge time of 72 hours is required to recharge the battery to the maximum charged condition assuming battery has completely discharged.

3.13.5 Spacecraft Operational Life

3.13.5.1 Lunar Day: The spacecraft shall be capable of operation from sunrise to sunset in any standard mode of operation, including the use of the above standby mode to equalize the electrical power dissipated as heat and the thermal dissipating capability of compartments A and B. Subject to the thermal constraints of paragraph 4.10 the spacecraft shall be capable of completing, without supplemental solar panel power, approximately 3 hours of postlanding operations which include the postlanding engineering verification and one narrow-angle black and white television survey.

3.13.5.2 Lunar Night:

3.13.5.2.1 Night Landing - For the case of a lunar night landing, the spacecraft shall be capable of a minimum of 3 hours for SC-1 through SC-4 and 20 hours for SC-5 through SC-7 of post-touchdown survival (non-operating).

3.13.5.2.2 Day Landing - For the case of a lunar day landing, and assuming a fully charged battery at sunset, the spacecraft (excluding payload energy requirement effects) shall be capable of a minimum of 150 hours of non-operating post-sunset survival.

3.13.5.3 Desired Operating Life - It is a desired objective to achieve the capability of a 90 day period of operation.

3.13.6 Alpha Scattering Instrument (SC 5,6 and 7)

3.13.6.1 Initiation: The operation of the alpha scattering instrument shall be initiated by earth command to the spacecraft. The following control of instrument operation shall be provided by earth command:

1) Alpha Detector No. 1 ON
2) Alpha Detector No. 2 ON
3) Proton Detector No. 1 ON
4) Proton Detector No. 2 ON
5) Proton Detector No. 3 ON
6) Proton Detector No. 4 ON
7) Calibration ON
8) Alpha Detectors No. 1 and 2 OFF
9) Proton Detectors No. 1 and 2 OFF
10) Proton Detectors No. 3 and 4 OFF
11) Calibration OFF
12) Alpha Scattering Power ON
13) Alpha Scattering Power OFF
14) Alpha Scattering Heater Power ON
15) Alpha Scattering Heater Power OFF
16) Deploy to background count position
17) Deploy to lunar surface

3.13.6.2 Timing (Alpha Scattering)

3.13.6.2.1 In the Typical sequence of operation the following will be completed: Standard sample count (3 hours), background count (3 hours), initial lunar surface count (6 hours), and data accumulation count (24 hours). The total interrupted time in either the standard sample or background count shall not exceed 30 minutes. The total interrupted time during the initial lunar surface count shall not exceed one hour. The time from initiation of the standard sample count and the initiation of the lunar surface count shall not exceed 7 1/2 hours.

3.13.6.2.2 A high quality television picture shall be obtained of the lunar surface (to which the instrument is to be deployed) and the instrument in the background position prior to deployment to the lunar surface.

3.13.6.2.3 Measurements of the sensor and electronics temperatures shall be obtained each 30 minutes of instrument operation.

4.0 SPACECRAFT PERFORMANCE

4.1 General

4.1.1 Functional Block Diagram*

A simplified block diagram of the Surveyor spacecraft is shown in Figure 4-1. This section (4.0) specifies the required system performance for these subsystems.

4.1.2 Weight and Inertia Control

4.1.2.1 The total "injected" weight and "payload" weight shall be as specified in JPL Specification 30240.

4.1.2.2* On the basis of present trajectory computations, the weight history may be assumed as stated in the HAC document titled Surveyor Spacecraft Monthly Performance Assessment Report.

4.1.2.3 For trajectory computations, the actual launch weights of each spacecraft shall be known to within 0.25 percent at least 120 days before the first possible launch day.

4.1.2.4 Minimum S/C weight at landing: 587 pounds.

4.1.2.5 Maximum S/C weight at landing: 645 pounds.

4.1.2.6 Maximum S/C dry landed weight: 605 pounds.

4.1.2.7 Minimum S/C weight at first acquisition of the range-velocity descent trajectory segments (see 4.4.4.2): 652 pounds.

4.1.2.8 Maximum S/C weight at first acquisition of the range-velocity descent trajectory segments (see 4.4.4.2): 715 pounds.

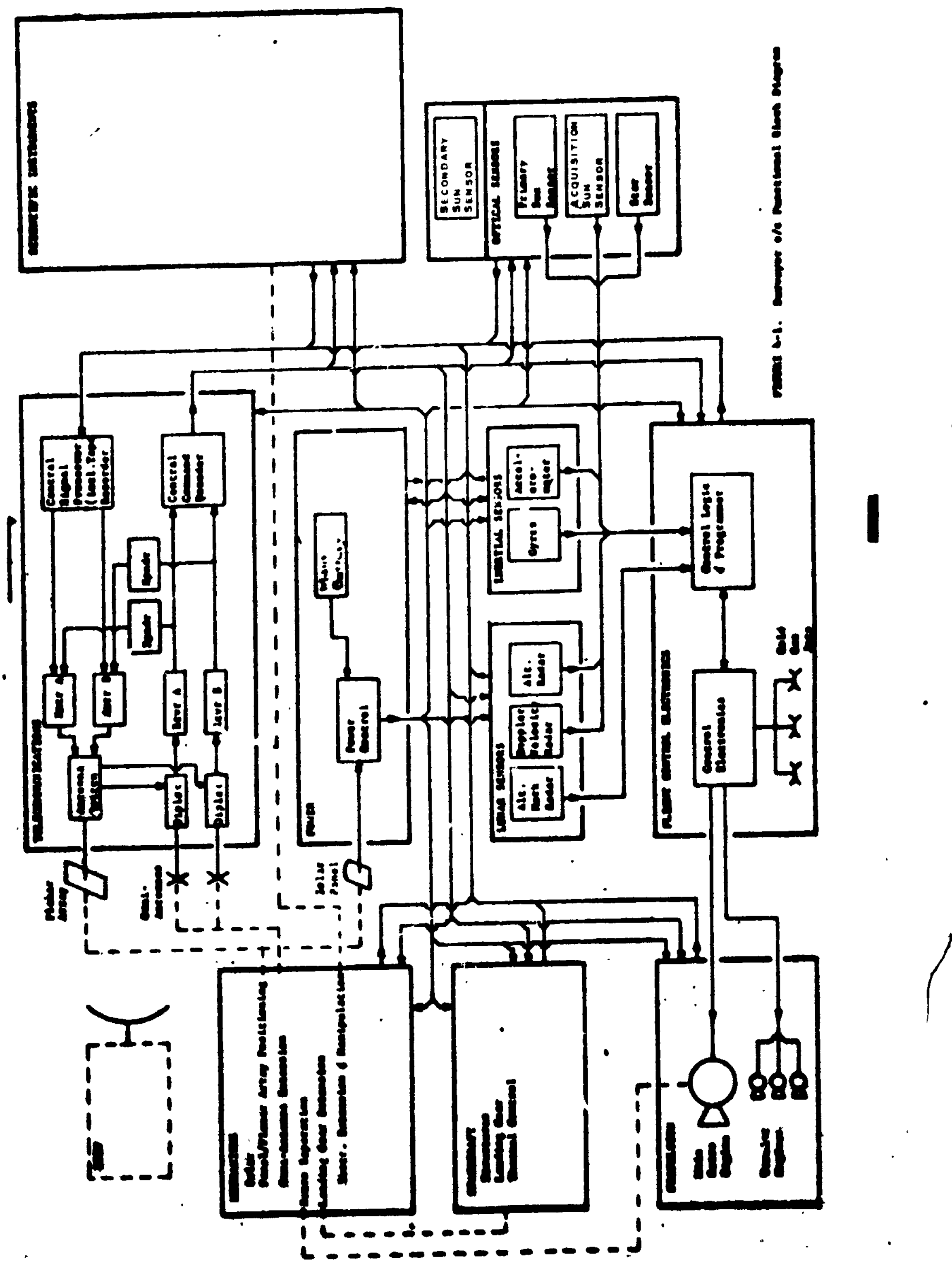


FIGURE 4-1
SURVEYOR S/C FUNCTIONAL BLOCK DIAGRAM

SURVEYOR LABORATORY HUGHES AIRCRAFT COMPANY SPACE SYSTEMS DIVISION	NUMBER 224510 REVISION H — MODEL SC PAGE 47 OF 177 PAGES

4.1.2.9 Ballast

The spacecraft design shall be such that up to 30 pounds of ballast can be added to the retro case in steps of no more than 3/4 pound to decrease the main retro deceleration for those cases in which the injection energy and the lunar approach velocity are low. This ballast shall be added to the retro case such that the moment generated by the added ballast weight with respect to the center of the retro mounting hole circle shall be less than 0.3 inch-pounds.

4.1.2.10 Moments of Inertia and Products of Inertia

These values shall be bounded by the following maxima and minima for the S/C throughout the landing phase from just prior to retro ignition with a full load of vernier fuel to just prior to touchdown with all vernier fuel gone.

I_x - between 116 and 229 slug ft^2

I_y - between 124 and 239 slug ft^2

I_z - between 119 and 224 slug ft^2

I_{zx} - less than 10 slug ft^2

I_{zy} - less than 10 slug ft^2

I_{yx} - less than 10 slug ft^2

For the S/C in the landing configuration just after retro separation (assuming all midcourse vernier fuel and the nominal vernier fuel required during retro burning and separation have been used), the actual moments of inertia shall be known for each S/C to ±5 percent for I_x and I_y, and ±10 percent for I_z.

4.1.2.11 Radius of Gyration

The radius of gyration for the spacecraft in the landing configuration of 4.1.3.1.3 shall be between 28 and 32 inches with respect to the X and Y axes.

4.1.3 Center of Gravity and Alignment Requirements

The basic alignment plane of th S/C is that plane defined by the column base fitting of the S/C base frame. The perpendicular to that plane passing through the geometrical center of the S/C optical tooling fixtures is the S/C roll axis as defined in Section 1.3. In the following requirements, the overall system requirement is first given followed by further breakdowns where applicable.

4.1.3.1 Spacecraft Horizontal c.g.

4.1.3.1.1 Launch Configuration

The c.g. of the total S/C, including the main retro engine, full load of vernier fuel, nitrogen gas, and helium gas, when in the launch configuration shall be within 1.00 inches of the S/C roll axis in a direction perpendicular to the roll axis. For SC-3 and SC-4 only, the center of gravity shall be within the envelope depicted in Figure 4-2.

4.1.3.1.2 Retro Configuration

The total c.g. of the S/C including main retro engine, vernier fuel, nitrogen gas, planar array and solar panel in landing condition, and excluding the altitude marking radar; shall be within 1.00 inches of the S/C roll axis throughout the main retro burning phase.

4.1.3.1.3 Landing Configuration

The c.g. of the S/C, less the main retro, vernier fuel, and nitrogen gas, when in the landing configuration shall be within 1.00 inches of the S/C roll axis in a direction perpendicular to the roll axis. The actual position of this c.g. shall be known for each S/C to within ±0.1 inch (2) for post-flight analysis purposes.

4.1.3.2 Spacecraft Vertical c.g.

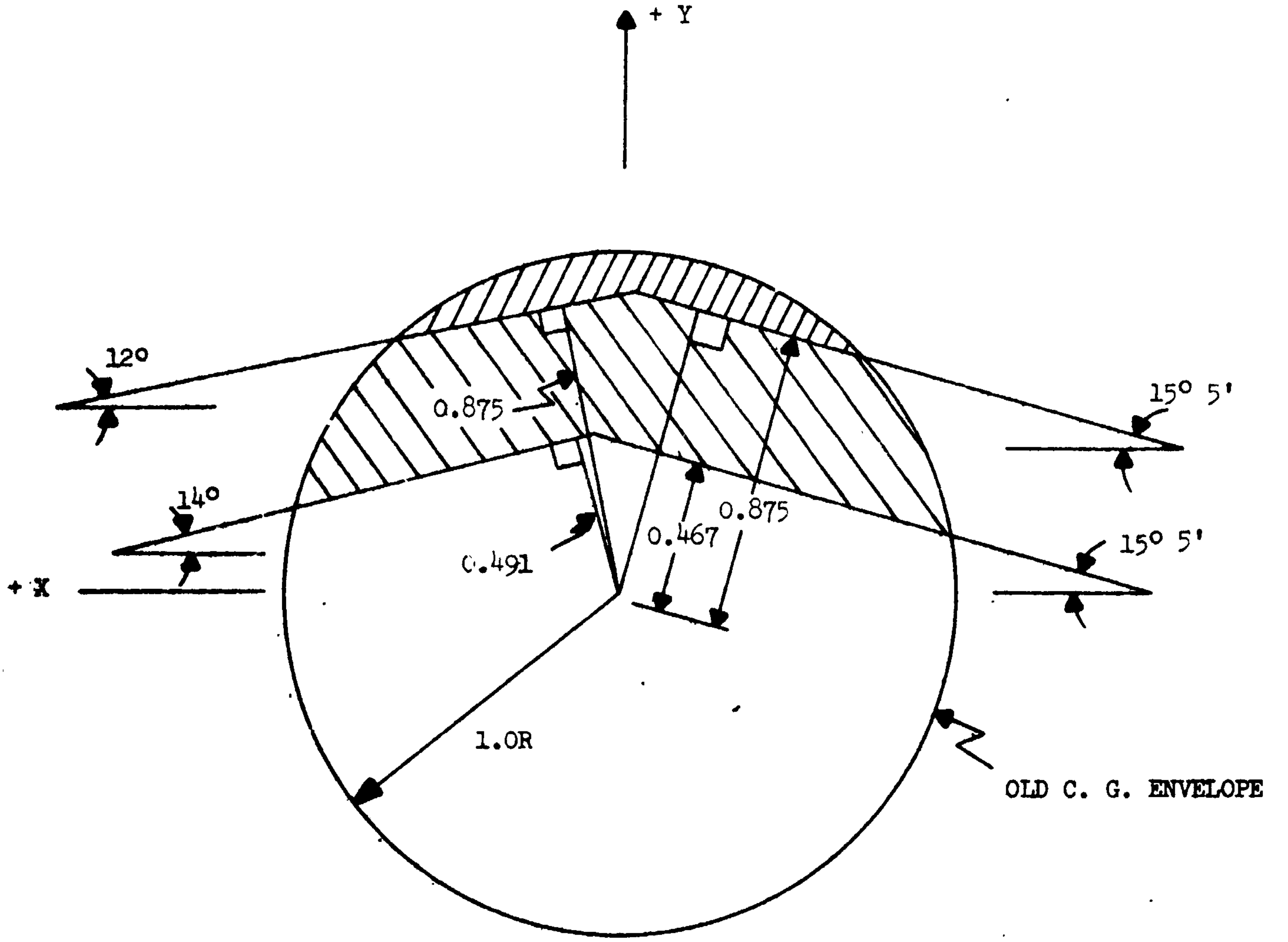
The c.g. of the S/C when in the landing configuration (as defined in 4.1.3.1.3 above) shall not be greater than 20.4 inches nor less than 16.9 inches above the S/C XY plane and this distance shall be known for each S/C to ±0.1 inch (2) (desired) or 0.25 inch (2) (required) for post-flight analysis purpose.

4.1.3.3 Offset Between the Retro Engine Thrust Vector and the Combined S/C-Main Retro c.g.

Throughout main retro burning for all expected environmental and spacecraft conditions, the perpendicular separation between the actual retro thrust vector and the combined S/C-main retro c.g. shall not exceed 0.18 inch (1). This separation is apportioned between the various contributors as follows:

4.1.3.3.1 Initial Alignment

The main retro engine shall be installed such as to minimize the offset between the retro engine thrust vector and the combined S/C-retro center of gravity at retro ignition.



Zone of main retro interference, mirrors vertical (center-of-gravity shall not fall within this area).

Mirrors are adjustable into this area if main retro mounting permits.

LAUNCH CONFIGURATION CG LIMITS

FIGURE 4-2

4.1.3.3.1.1 Install tion

The offset between the predicted location of the retro thrust vector at retro ignition and the predicted combined S/C-retro c.g. under alignment conditions shall be less than 0.005 inches.

4.1.3.3.1.2 Spacecraft Center of Gravity

The error in predicting the location of the spacecraft center of gravity under the following alignment conditions shall be less than 0.015 inches (1):

a) The S/C is in the retro descent configuration (e.g., marking radar ejected, planar array and solar panel in landing conditions, landing legs and omni antenna extended).

b) Nitrogen gas weight equal to the loaded weight minus the expected usage prior to retro ignition.

c) Vernier propellant used is 1/2 of that required for a 3σ midcourse maneuver plus 1.1 second pre-retro operation.

d) Helium gas distribution corresponding to that required to expel vernier propellant to the vernier propellant alignment condition.

e) Planar array at nominal landing position selected for the particular launch.

4.1.3.3.1.3 Retro Engine Assembly Center of Gravity

As given in 4.2.1.11

4.1.3.3.1.4 Initial Retro Engine Thrust Vector Location

As given in 4.2.1.11

4.1.3.3.1.5 Planar Array Position

To allow for a change in landing site following launch, the shift in the combined S/C-retro center of gravity from the alignment condition produced by corresponding change in planar array position within the angular range as given in paragraph 4.8.1 3.3 shall be less than 0.005 inches.

4.1.3.3.1.6 Combined Center of Gravity Ballast Limitations

The maximum ballast that may be added to the spacecraft landing pads to align the spacecraft/retro combined center of gravity to the retro nozzle centerline is listed below for each leg:

Leg number 1	1.5 pounds
Leg number 2	1.5 pounds
Leg number 3	0.25 pounds

4.1.3.3.2 <u>Change in Offset During Retro Period</u>

The change in retro thrust vector to combined center of gravity offset shall be controlled by the following requirements:

4.1.3.3.2.1 <u>S/C Deformations Due to Heat and Thrust Load</u>

As given in 4.1.3.16.1.

4.1.3.3.2.2 <u>Vernier Fuel System c.g. Excursions</u>

As given in 4.1.3.6.4.

4.1.3.3.2.3 <u>Retro Engine c.g. Excursions During Burning</u>

As given in 4.2.1.12.

4.1.3.3.2.4 <u>Retro Engine Thrust Excursions During Burning</u>

As given in 4.2.1.13

4.1.3.4 <u>Retro Engine Thrust Vector Angular Alignment:</u>

Throughout main retro burning for all expected environmental and S/C conditions, the angle between the actual main retro thrust vector and the sensor group roll axis boresight shall not be greater than 0.4 degrees (1). This specification shall include the following:

4.1.3.4.1 <u>Retro Installation:</u>

The installation of the retro engine to the spacecraft shall be controlled such that a line perpendicular to the retro reference plane (see Section 4.2) shall be parallel to the spacecraft roll axis (see Section 1.3), within 0.05 degree (1).

4.1.3.4.2 <u>Sensor Group Installation</u>

As given in 4.1.3.7

4.1.3.4.3 <u>Sensor Group Alignment Excursions Due to Heat and Thrust and Vibration Loads</u>

As given in 4.1.3.7.3

4.1.3.4.4 Retro Engine Thrust Alignment

As given in 4.2.1.11.

4.1.3.4.5 Retro Engine Thrust Angular Excursions During Burning

As given in 4.2.1.13 and 4.1.3.16.2.

4.1.3.5 Vernier Engine Thrust Alignment

4.1.3.5.1 The vernier engines including the swiveled engine when at its mechanically centered position, shall be installed so that the following overall alignment requirements can be met under the stated conditions throughout the life of the engine.

4.1.3.5.1.1 With Main Retro Off

With the main retro off and the vernier engines at any thrust level, the angle between the actual thrust vector of any one engine and the S/C roll axis shall not exceed 0.70 degree(1).

4.1.3.5.1.2 With the Main Retro On

Throughout main retro burning with the vernier engines at mid-thrust, the angle between the actual thrust vector of any one engine and the S/C roll axis shall not exceed 0.55 degree(1).

4.1.3.5.2 These overall requirements (4.1.3.5.1) are made up of the following contributors.

4.1.3.5.2.1 Installation Alignment

The vernier engine alignment angle is defined as the angle formed by rotating the vernier engine about its trunnion axis such that the nozzle centerline moves from a position which is parallel to the spacecraft roll axis to the aligned position. A positive angle signifies that the vernier engine is rotated such as to move the nozzle towards the engine mounting bracket. The alignment angle shall be +0.15 degree for all engines. In a direction normal to rotation about the trunnion axis, the engine centerline shall be aligned parallel to the spacecraft roll axis. The angular alignment tolerance for all engines in both of the above directions shall be 0.075 degrees (1), except for the alignment angle of engine 1 which shall be 0.10 degree (1). The alignment tool weight shall be 11 ± 1 pounds. The supported weight on the bracket shall be 8.49, 6.80, 6.79 pounds for engines 1, 2, 3, respectively. The supported weight shall be within 0.2 pound of the above values.

4.1.3.5.2.2 Vernier Engine Mounting Bracket Bending Constant

As given in 4.1.3.16.6.

4.1.3.5.2.3 Vernier Engine Thrust Vector Excursions

As given in 4.2.2.11.

4.1.3.5.2.4 Environmental Effects· The shift in angular alignment of the vernier engines from the initial preflight alignment condition due to flight environment excluding bracket bending resulting from acceleration and thrust loads shall be less than 0.1 degree (2).

4.1.3.6 Vernier Engine System to Sensor Group Roll Axis Alignment

In this section, the vernier engine system shall include all nozzles, control valves fuel tanks, fuel lines, helium tank, and helium lines as installed on the S/C.

4.1.3.6.1 Moment Arm Length

For each spacecraft, the distance from the spacecraft roll axis to each of the vernier engine nozzle centerlines shall not deviate from the mean of the three distances for that spacecraft by more than 0.10 inch (2).
The minimum moment arm length shall be 37.7 inches.

4.1.3.6.2 Vernier Roll Control Gimbal Axis

Given a theoretical line passing through the vernier thrust axis-gimbal axis junction and also passing through and perpendicular to the S/C roll axis, the included angle between this line and the vernier roll control gimbal axis shall be no greater than 1.5 degrees (1).

4.1.3.6.3 Vernier Propellant Loading Accuracy

The specified propellant load for each fuel or oxidizer tank is one third the total specified load of fuel or oxidizer. The indicated propellant weight loaded in each of the six propellant tanks shall be within 0.25 pounds (2) of the specified loading for that tank. The total error in measurement of the actual propellant loaded in each tank shall be less than 0.1 pourd(2). The positive tolerance on propellant tank loading shall not reduce the minimum usable capacity of the vernier propellant system.

4.1.3.6.4 Vernier Propellant c.g. Excursions

4.1.3.6.4.1 Vernier Propellant c.g. Loaded Condition

When the vernier propellant tanks are filled to the specified loading, the c.g. of the vernier propellant shall be within 0.1 inch of the S/C roll axis.

4.1.3.6.4.2 Vernier Propellant c.g. During Main Retro Phase

Throughout main retro burning, the vernier propellant center of gravity shall shift in such a direction as to shift the combined spacecraft-retro c.g. toward the retro thrust vector.

4.1.3.6.4.3 Vernier Propellant c.g. During Vernier Descent Phase

The S/C c.g. shall not shift more than 0.2 inch(1) in a direction parallel to the XY plane during the vernier descent phase due to uneven propellant distribution.

4.1.3.7 Flight Control Sensor Group Support Structure Alignment

This group includes the primary sun sensor, the Canopus sensor, the roll, pitch and yaw attitude gyros, and accelerometers.

4.1.3.7.1 The sensor group support structure roll axis shall be aligned to the vehicle roll axis within 0.10 degree(1).

4.1.3.7.2 The sensor group support structure pitch axis shall be aligned to the vehicle pitch axis within 0.40 degree(1).

4.1.3.7.3 Angular alignment uncertainty of the sensor group to the vehicle roll axis due to acceleration and vibration loads and heat loads shall not exceed 0.20 degree(1).

4.1.3.8 Canopus Sensor Alignment

The Canopus sensor boresight axis shall be aligned to the sensor group support structure pitch axis such that the overall pointing specification of 4.3.1.2 can be met. Further, it shall be possible to position the Canopus sensor field-of-view in the plane of the sensor group pitch-roll axes to the accuracy and over the limits specified in 4.5.4.2.

4.1.3.9 Gyro Alignment

Each of the three gyros shall be aligned to their respective sensor group axes such that the specifications of 4.3.1 can be met.

4.1.3.10 Thrust Control Accelerometer Alignment

The accelerometer sensitive axis shall be aligned along the sensor group roll axis such that the thrust control requirements of 4.4 can be met.

4.1.3.11 Altitude Marking Radar

The antenna axis shall be within 0.50 degree[1] of the vehicle roll axis provided the radar attach points shall be within 0.25 degree[1] of proper orientation, following exposure to all pre-retro S/C environments.

4.1.3.12 Radar Altimeter and Doppler Velocity Sensor

During vernier descent, the radar altimeter antenna axis (RF beam) shall be aligned within 0.40 degree[1] of the vehicle roll axis; and the doppler velocity sensor antenna axes (RF beam) shall be aligned within 0.40 degree[1] of the desired directions. During main retro burn, the doppler velocity sensor antenna axes shall be aligned within 1.0 degrees[1] of the desired directions. (See 1.3 for definition of antenna axes.)

4.1.3.13 Antenna/Solar Panel Positioner

The antenna/solar panel positioner shall be installed such that the requirements of 4.8.1.3.4 can be met.

4.1.3.14 Primary Sun Sensor Alignment

The primary sun sensor shall be installed on the sensor group support structure such that the overall pointing specification of 4.3.1.1 can be met.

4.1.3.15 Secondary Sun Sensor Alignment to Primary Sun Sensor

The secondary sun sensor is nominally used after touchdown but may also be used as a backup for the acquisition sun sensor (See 4.5.3). To ensure this backup capability, the secondary sun sensor roll, pitch and/or yaw axes must be aligned to the primary sun sensor corresponding axes, when the solar panel is in its transit position, to within ±2.0 degrees[1] in each axis. This total alignment specification is a result of the following contributing specifications:

4.1.3.15.1 Secondary Sun Sensor Null Accuracy

As given in 4.5.2.2.

4.1.3.15.2 Secondary Sun Sensor Mounting to Solar Panel

As given in 4.8.2.7.

4.1.3.15.3 Solar Panel Mechanism Assembly

4.1.3.15.4 Vehicle Structures

In providing for the planar array-solar panel mast and the transit roll lock, vehicle structures shall contribute less than ±0.5 degree(1) to the total alignment error.

4.1.3.15.5 Primary Sun Sensor Alignment

As required by 4.1.3.14 and 4.5.1.4

4.1.3.16 Spacecraft Structural Deformation

4.1.3.16.1 Retro Nozzle Centerline to S/C c.g.

For all expected environmental conditions, the displacement between the retro nozzle centerline and the combined S/C-retro engine c.g. shall change no more than ±0.07 inch(1) from the static no thrust position during main retro thrusting.

4.1.3.16.2 Retro Nozzle Centerline to S/C Roll Axis

For all expected environmental conditions, the angle between the retro nozzle centerline and the S/C roll axis shall change by not more than 0.15 degree(1) from the static no thrust condition due to main retro thrusting.

4.1.3.16.3 Sensor Group Roll Axis to S/C Roll Axis

As given in 4.1.3.7.3.

4.1.3.16.4 Vernier Engine Angular Alignment

Included in 4.1.3.5.

4.1.3.16.5 Antenna/Solar Panel Positioner Static Deflections

As given in 4.8.1.3.5 and 4.8.2.3.5.

4.1.3.16.6 Vernier Engine Mounting Bracket

The vernier engine bracket bending constant shall be less than 0.50 degrees per 100 pounds[2] of load applied along the vernier engine centerline. Expected value and standard deviation of bracket stiffness is required one month before vernier engine alignment.

4.1.3.16.7 Omni-Antenna Dynamic Deflections

(a) During engine burning phases, both retro and vernier as well as vernier only, the omni-antenna motion at the end of the boom shall not exceed a magnitude of 5 cm peak-to-peak and shall not have a fundamental frequency component of greater than 30 cps.

(b) During a normal lunar landing (i.e., in specification touchdown conditions) the omni-antenna motion at the end of the boom shall not exceed a magnitude of 20 cm peak-to-peak and shall not have a fundamental frequency component of greater than 10 cps.

4.1.3.16.8 Planar Array Dynamic Deflections

During main retro descent, the planar array translational motion shall not exceed a magnitude of 5 cm peak-to-peak and shall not have a fundamental frequency component of greater than 30 cps.

4.2 Propulsion

4.2.1 Main Retro Engine

4.2.1.1 Useful Total Impulse (See 1.3)

See classified addendum. The allowable variation in useful total impulse from the nominal value is apportioned below.

4.2.1.1.1 Propellant Loading Accuracy

As given in Section 4.2.1.5.

4.2.1.1.2 Unusable Propellant

Weight of propellant remaining in the engine at the end of retro interval (See 1.3) shall be less than 1 pound[2].

4.2.1.1.3 Useful Specific Impulse Variation (See 1.3)

As given in 4.2.1.2.

4.2.1.2 Useful Specific Impulse (See 1.3)

See classified addendum. The nominal value of useful specific impulse shall be specified not later than four months prior to scheduled launch date.

4.2.1.3 Thrust Levels

The maximum instantaneous thrust over the operating temperature and at vacuum conditions shall not exceed 10,000 pounds. The minimum instantaneous thrust over the temperature range at vacuum conditions shall not be less than 8000 lbs during the time interval from 10 seconds after retro ignition until the beginning of the thrust decay.

4.2.1.4 Thrust Time Profile (See 1.3)

See classified addendum.

4.2.1.5 Propellant Loading

Retro shall be capable of being loaded to any propellant weight between 1197 and 1300 pounds. Allowable error in propellant loading is ±0.3 per cent(2) of the specified propellant weight. Actual propellant weight shall be known after loading to ±0.1 percent(2). Propellant load for each launch shall be specified not later than three months prior to scheduled launch date.

4.2.1.6 Propellant Temperature

The average grain temperature at time of operation shall be 55°F ±20°F(2). The minimum temperature at any point within the propellant shall be 10°F. The maximum error in prediction of the average grain temperature at retro ignition shall be 20°F(2).

4.2.1.7 Ignition Time (See 1.3)

The nominal engine ignition time (see 1.3) shall be 0.30 second for any temperature within the range of +20°F to +70°F. The variation on ignition time shall not exceed 0.050 second(2).

4.2.1.8 Retro Operate Interval (See 1.3)

See classified addendum.

4.2.1.9 Retro Inert Weight Change During The Retro Interval

Four months prior to launch, or earlier, the retro inert weight loss from ignition to separation shall be known to within ±30 percent[2].

4.2.1.10 Thrust Decay

The time increment measured from a vacuum thrust level of 3,500 pounds to the point where thrust has decayed to less than 30 pounds shall not exceed 12.0 seconds.

4.2.1.11 Center of Gravity and Thrust Alignment:

The reference plane for the retro engine is determined by three points which are the intersections of the centerline of the cylindrical hole in the conical bushings and the plane of the machined surface on the retro mounting brackets. The angle between the nozzle centerline and the perpendicular to the reference plane shall not exceed 0.02 degree. The center of the reference plane is defined as the center of the circle containing the above three intersections. The maximum deviation between both the loaded and unloaded retro center of gravity projected perpendicularly to this plane and the center of the reference plane shall be less than 0.030 inch. The distance between the nozzle centerline and the center of the reference plane as measured in the reference plane shall be less than 0.007 inch

4.2.1.12 Center of Gravity Excursion

The engine c.g. shift during burning shall not exceed 0.03 inch[1] in the reference plant.

4.2.1.13 Thrust Excursion+

Actual thrust vector shift in the reference plane during burning shall be no greater than 0.04 inch[1] and the angular excursion between the actual thrust vector and the initial position of the nozzle centerline shall not exceed 0.2 degree[1].

+Nozzle centerline and actual thrust vector (see 1.3) are assumed to be coincident. If the actual thrust vector deviates from the nozzle centerline, then the allowances with respect to the nozzle centerline must be correspondingly reduced to satisfy the overall tolerances.

4.2.2 Vernier Engine System

4.2.2.1 Specific Impulse

Vernier engine specific impulse shall be equal to or greater than the following nominal values at maximum, mid-range, and minimum thrust levels.

	Nominal	
Maximum thrust	287	± 6 sec.(2)
Mid-range thrust	278	± 6 sec.(2)
Minimum thrust	273	± 6 sec.(2)

4.2.2.2 The nominal vernier engine oxidizer-fuel (O/F) ratio shall be 1.5 ± 0.1(2).

4.2.2.3 The minimum useable propellant capacity of the vernier engine system shall be 178.35 pounds at 100°F.

4.2.2.4 Total Propellant Allowances

On the basis of present trajectory computations, the breakdown of propellant weight allowances may be assumed as stated in the HAC document titled Surveyor Spacecraft Monthly Performance Assessment Report.

4.2.2.5 Vernier Propellant Loading Accuracy

Vernier propellant loading accuracy shall be as specified in 4.1.3.6.3.

4.2.2.6 Startup Impulse

Startup impulse shall be such that the requirements of paragraphs 3.6, 3.8.3, and 3.9 are met.

4.2.2.7 Shutdown Impulse

The nominal overall shutdown impulse for each engine (including engine, throttle valves, switches and power supply), when operating at 77 pounds thrust, shall be less than 5 lb-sec. The difference in shutdown impulse between any two engines when operating at the same thrust within 4 pounds shall be no greater than 0.66 lb-sec(2). The nominal value of shutdown impulse shall be determined prior to spacecraft launch.

4.2.2.8 Hysteresis

For complete and continuous thrust excursions at rates in which system dynamic lags are not effective, the width of all thrust versus command loops shall be less than 3 pounds or 15 percent of the command thrust excursion, whichever is larger.

4.2.2.9 Small Signal Frequency Response

For a sinusoidal differential thrust command whose amplitude is 8.7 pounds peak-to-peak about any level such that the instantaneous thrust is greater than 30 pounds and less than 104 pounds, and frequency is 5 cycles per second, the ratio of thrust excursion to peak-to-peak thrust command shall be greater than 0.85 times the ratio of average thrust to average thrust command. The phase shift between sinusoidal component of thrust command and sinusoidal component of thrust produced shall be less than 32 degrees.

4.2.2.10 Large Signal Response

The time for the engine thrust to change from a value less than 35 pounds to greater than 104 pounds, or from a value greater than 98 pounds to less than 30 pounds, in response to a step change in thrust command shall be less than 0.120 seconds.

4.2.2.11 Thrust Vector Angular Alignment

The actual thrust vector of any vernier engine (see 1.3) shall not vary more than 0.33 degree[1] prior to retro burnout (3.5 g point), nor more than 0.45 degree[1] for the duration of the mission, from the engine geometrical centerline (as established by the engine alignment tool prior to launch) throughout engine life.

4.2.2.12 Engine Thrust

Each engine shall be capable of operation at a minimum thrust of no more than 30 pounds, a maximum thrust of not less than 104 pounds, and varying thrust levels between these extremes without restriction, throughout the use of the full load of usable fuel.

4.2.2.13 Throttle Ratio

Each engine shall have a throttling capability of at least 3.47 to 1 (greater throttling ratios are strongly desired).

4.2.2.14 Engine Thrust Command

For a given thrust command signal into the vernier engine system, the actual thrust shall equal the commanded thrust with the following accuracies:

Commanded Thrust	Actual Thrust
30 lbs	30 lbs $^{+5}_{-6}$ lbs
67 lbs	67 lbs $^{+3}_{-3}$ lbs
104 lbs	104 lbs $^{+16}_{-5}$ lbs

4.2.2.15 Engine Start Capability

Excluding any pre-launch operations, the engines shall be capable of at least 4 starts during flight. The time between a subsequent ignition of the vernier engines and the termination of the previous ignition of the engines shall be at least one hour.

4.3 Attitude Control

Vehicle attitude is controlled by combinations of three attitude sensing instruments and two power or vehicle torquing devices. Control accuracies are specified in terms of the difference between actual and desired position of a given spacecraft orthogonal coordinate system. Coordinate systems used in this section are "sensor group" and "spacecraft". Their definitions and relationships can be found in Section 1.3.

4.3.1 Steady-State Control Accuracies

4.3.1.1 Sun Reference with Gas Jet Torquing

The sensor group roll axis shall be held to within 0.20 degree of the Sun-S/C line, plus a ±0.30 degree limit cycle.

4.3.1.2 Canopus Reference with Gas Jet Torquing

The angle between the sensor group roll-pitch plane and the projection of the Canopus – S/C line on the sensor group pitch-yaw plane shall be held to within 0.20 degree of zero, plus a ±0.30 degree limit cycle.

4.3.1.3 Inertial Reference with Gas Jet Torquing

Assuming the items listed below, direction of each axis of the sensor group coordinate system shall be known and controlled with respect to the zero direction (as defined below) within 0.22 degree plus a limit cycle amplitude about this point of ±0.30 degree.

a) The zero inertial reference direction for the sensor group coordinate system is that direction in space which exists when control reference is switched from optical to inertial.

b) Command rotation about any axis is ≤ 180 degrees.

c) Gyro drift is not included in the accuracy requirement listed above.

4.3.1.4 Inertial Reference with Vernier Engine Torquing

When the previous control mode is inertial reference with gas jet torquing, the sensor group coordinate system position shall be known and controlled with respect to the zero position (as defined in 4.3.1.3a) to within 0.10 degree(2), gyro drift excluded, in all axes when the main retro is off and to within 0.30 degree(2) in the pitch and yaw axes when the main retro is on. With the main retro on, the error in roll may increase to 0.5 degree(2) (desired) or 1.0 degrees(2) (required), on the assumption that no roll disturbing moment will exist as a result of vortex action within the retro engine.

(a) Gyro non g sensitive drift shall be less than 1 degree per hour (2).

(b) Gyro g sensitive drift shall be less than 2 degrees per hour per g(2).

(c) Gyro anisoelastic drift shall be less than 0.02 degree per hour per g^2(2).

(d) Gyro elastic restraint drift shall be less than 0.6 degree per hour per degree (2).

4.3.2 Attitude Rates

4.3.2.1 During precision maneuvers for midcourse and retro pointing, the steady state S/C attitude rate shall be 0.5 ± 0.0011 degree per second.

4.3.3 Moment Control Capability

4.3.3.1 Vernier Engine System

4.3.3.1.1 Pitch and Yaw

The attitude control system must be capable of maintaining a stable attitude about any axis through the combined S/C-retro center of gravity, displaced from the roll axis per paragraph 4.1.3.1 and parallel to the pitch-yaw plane, with the following minimum control moment capability:

a) 1890 pound-inches when the total thrust is controlled to 200 pounds.

b) 2065 pound-inches in the 150 or 200 pound command mode, by allowing the total thrust to increase or decrease to that required to provide the moment required.

4.3.3.1.2 Roll Channel

The vernier roll engine at 30 pounds thrust shall produce at least ±90 pound-inches[2] of moment control about the vehicle roll axis. The 99% probability steady state roll actuator angle required to maintain roll attitude in the presence of maximum allowable moment disturbances shall be less than ±4.5 degrees. The minimum roll actuator stop limit, in either direction, shall be 5.4 degrees.

4.3.3.2 Cold Gas Jet System

The control moment provided by the cold gas jet system shall be:

Roll - Greater than 4.00 inch-pounds
Pitch - Greater than 4.25 inch-pounds
Yaw - Greater than 7.00 inch-pounds

4.3.4 Attitude Maneuver Capability of Cold Gas Jet System

Listed below are those maneuvers considered to be a requirement for a standard lunar mission. The gas-jet attitude control system must have the necessary propellant to complete these maneuvers with a 3σ probability. In addition, the gas jet attitude control system must have approximately a 100 percent safety factor to take care of non-standard conditions.

a) Rate mode : 30 minutes

b) Rate absorption : 6 deg/sec

c) Optical mode : 64 hours

d) Inertial mode : 2 hours

e) Sun acquisition : 1 maneuver

f) Star Acquisition : 1 maneuver

g) Roll : 6 maneuvers

h) Yaw (or pitch) : 5 maneuvers

i) Star verification : 4 maneuvers

4.3.5 Attitude Maneuver Rates With Vernier Engine Control System

At least 5.5 deg/sec in pitch and yaw axes.

4.3.6 Attitude Control Dynamic Response

4.3.6.1 Start Up Transients

Attitude transients due to vernier engine ignition shall be reduced to a level wherein the accuracy requirements of 4.3.1.4 can be met within 1 sec after engine ignition.

4.4 Thrust Control

Vernier engine thrust is controlled by several independent reference sources as described in the following modes of operation:

4.4.1 Midcourse Velocity Correction

In this mode, the vernier engine thrust shall be servoed to provide a nominal fixed vehicle acceleration of 3.22 ± 0.04 ft/sec^2 for a radio commanded time interval variable from zero to 51.0 seconds.

4.4.2 Main Retro Phase

In this mode, the average vernier thrust shall be maintained at a constant level by the thrust control system while individual engines are differentially throttled by the attitude control system to counteract main retro thrust offset moments. Average total vernier thrust shall be radio commandable to one of two values: 200 pounds ±10 percent(2) or 150 pounds ±10 percent(2). When the 150 pound thrust level is commanded, severe attitude control commands to individual engines may cause one or more engines to saturate and thus raise the average thrust level above 150 pounds.

4.4.3 Main Retro Separation

This mode begins at the 3.5g acceleration level during main retro thrust decay (see Section 3.9) and ends 2 seconds after the retro separation bolts have been exploded. To facilitate clean separation of the main retro from the S/C, average total vernier thrust shall be maintained at a minimum constant level of 280 pounds(2).

4.4.4 Vernier Descent Phase

This phase is divided into three modes of vernier engine thrust control. They are: (a) acceleration control, (b) velocity control by range reference, and (c) constant velocity control. Listed in order below are the specifications covering these modes of control:

4.4.4.1 Acceleration Control

Following the main retro separation mode and prior to altitude reliable operate signals, the vernier engine thrust shall be servoed to maintain a constant thrust to mass ratio of 4.88 ± 0.14 ft/sec^2. The minimum S/C weight during this interval shall be as specified in 4.1.2.7.

4.4.4.2 Range Reference Velocity Control

In this mode, the S/C will be commanded to follow a four segment z-axis velocity versus slant range descent trajectory. The segment end points (listed below) fall near a constant thrust to mass ratio descent trajectory. When the steady state is reached on the first segment, the error between the commanded velocity and the measured velocity shall be less than 2 feet per second (2).

When the S/C is above the segmented trajectory and the velocity control is saturated, the S/C thrust to mass ratio shall be as specified in paragraph 4.4.4.1. When the S/C is below the segmented trajectory and the velocity control is saturated, the actual S/C thrust to mass ratio shall be 12.51 ± 0.19 ft/sec^2.

It should be assumed for the purpose of meeting this specification that: (a) the S/C thrust axis will become aligned to the velocity vector prior to making contact with the segmented trajectory, (b) sensor limitations are not exceeded, and (c) the S/C maximum weight at the time of intercept with the command trajectory is as specified in 4.1.2.8.

SEGMENT END POINTS

Segment	Lower End		Upper End	
	Range (Ft)	Velocity (Ft/Sec) $\pm 3\sigma$	Range (Ft)	Velocity (Ft/Sec) $\pm 3\sigma$
1	0	0 ± 0.5	327	$58 \pm 2\%$
2	327	$58 \pm 2\%$	1,050	$108 \pm 2\%$
3	950	$102 \pm 2\%$	13,160	$390 \pm 2\%$
4	13,160	$390 \pm 2\%$	45,200	$750 \pm 2\%$

4.4.4.3 Constant Velocity Control

See 3.10.1.5 The minimum weight during this mode is as specified in 4.1.2.4.

4.5 Optical Sensors

The flight control sensor group includes an acquisition sun sensor, a primary sun sensor, and a star (Canopus) sensor. The acquisition sun sensor affords sufficient attitude control accuracy to permit sun acquisition by the primary sun sensor. The primary sun sensor and the star sensor provide accurate attitude reference information both during coast phase and for initiation of attitude maneuvers. In addition, a secondary sun sensor, located on the solar panel, is used after lunar landing for positioning the solar panel during the lunar day; it may also be used before lunar landing as a back-up for the functions of the acquisition sun sensor.

4.5.1 Primary Sun Sensor

4.5.1.1 Field-of-View

The primary sun sensor field-of-view shall insure automatic sun acquisition after initial positioning with the accuracy afforded by using the acquisition sun sensor. In addition, the primary sun sensor field of view shall be such that lock-on and the required tracking accuracy can be maintained with the planar array in either its coast phase position or its terminal descent position.

4.5.1.2 Lock-on Signal

The primary sun sensor shall provide a signal whenever the sun is within its usable field-of-view.

4.5.1.3 Tracking Accuracy

When locked on and tracking the sun, the error signal accuracy shall be such that the over-all attitude control accuracy of 4.3.1.1 can be met.

4.5.1.4 Alignment

The alignment accuracy of the primary sun sensor on the sensor group support structure shall be such that the over-all attitude control accuracy of 4.3.1.1 can be met.

4.5.2 Secondary Sun Sensor

4.5.2.1 Field-of-View

The secondary sun sensor field-of-view shall be at least one hemisphere.

4.5.2.2. Accuracy

The secondary sun sensor shall be capable of providing an output which indicates, after landing, when the true S/C-sun vector is within ±0.75 degree(2) of the secondary sun sensor boresight axis. (This accuracy is more than adequate for use as a sun acquisition back-up before lunar landing.)

4.5.3 Acquisition Sun Sensor

4.5.3.1 Field-of-View

The acquisition sun sensor shall provide a clear, fan-shaped field-of-view, fixed in spacecraft coordinates and centered on the -X axis, covering at least ±3° from the X, Z plane over an angular region of ±90° in the X,Z plane.

4.5.3.2 Output

The acquisition sun sensor shall provide an on-off switching signal whose "on" state indicates correctly the presence of the sun within its field-of-view. This signal is used for those flight control attitude maneuvers required to permit sun acquisition by the primary sun sensor. The acquisition sun sensor output memory latch is telemetered.

4.5.3.3 Alignment

The installation tolerance between the acquisition sun sensor and the FCSG mounting interface shall be such that each of the three acquisition sun sensor axes (pitch, yaw and roll) shall be aligned within 0.5 degrees of the corresponding axes of the primary sun sensor.

4.5.4 Canopus Sensor

4.5.4.1 Field-of-View

The instantaneous angular field-of-view of the output signal axis (roll axis) shall be approximately 4.0 degrees. Field-of-view in the yaw (XZ) plane shall be enough (>5 degrees) to make adjustments of the sensor unnecessary over an eight-day launch period.

4.5.4.2 Position of the Field-of-View

The center of the field-of-view shall be positionable, prior to launch in the plane formed by the vehicle roll and pitch axes. In this plane, the angle from the roll axis to the center of the field-of-view shall be adjustable to any angle between 75° and 105° with an accuracy of ±0.5 degree (2).

4.5.4.3 Lock-on Signal

The sensor shall provide a constant output (separate from the control signal) when Canopus is within the field-of-view.

4.5.4.4 Discrimination

4.5.4.4.1 Mode I

The sensor shall provide upper and lower intensity discrimination levels such that Canopus will be detected and locked onto while all other celestial bodies within view are rejected.

4.5.4.4.2 Mode II

The sensor shall provide intensity sensitivity such that a complete star map of stars down to 2.8 magnitudes below Canopus (S4 response) can be made of those celestial bodies passing in view as the vehicle is rolled 360°. Additional sensitivity is strongly desired.

4.5.4.5 Accuracy

The Canopus error signal null accuracy and scale factor shall be such that the over-all attitude control accuracy of 4.3.1.2 can be met.

4.5.4.6 Alignment

The alignment accuracy of the Canopus sensor on the sensor group support structure shall be such that the over-all attitude control accuracy of 4.3.1.2 can be met.

4.6 Lunar Sensors

The Surveyor lunar sensors include the Altitude Marking Radar (AMR) and the Radar Altimeter and Doppler Velocity Sensor (RADVS). For purposes of the design and evaluation of these radars, it will be assumed that the radar reflecting properties of the moon are as described in 4.6.1, below.

4.6.1 Lunar Radar Reflectivity Model

For design, pre-flight evaluation, and post-mission analysis of the AMR and RADVS, the lunar radar reflectivity model shall be the revised Muhleman model in the form described herein.

4.6.1.1 Definition

That definition fund[illegible]ntal to radar analysis is the ratio of back-scattered power density to total incident power for a stated reflector. For an increment dA of area on the lunar surface, this ratio is defined herein as:

$$\frac{dP_s}{dP_i} = \eta \frac{F(\theta_i)}{4\pi}$$

where

dP_i = total power (watts) incident on dA.

dP_s = power spatial density (watts per steradian) back-scattered by dA.

θ_i = angle of incidence (off lunar vertical) of that incremental bundle of rays (by geometrical optics) containing dP_i and intercepted by dA.

η = radar reflectivity factor (dimensionless), as defined further in 4.6.1.2

$F(\theta_i)$ = angle-dependent function, as defined further in 4.6.1.4

4.6.1.2 Total Hemisphere Average Radar Cross-Section

The factor η is itself defined as the ratio of power back-scattered by the actual surface to that power which would be back-scattered if the surface scattering properties were lossless and isotropic.

This definition is such that the factor η is the ratio of the commonly defined effective radar cross-section of the total lunar hemisphere (as viewed from Earth) to the total lunar physical cross-section (area of the projected planar disc).

All currently available information indicates that the present best estimate of this factor is:

$$\eta = 0.075$$

This value shall be assumed as the nominal value.

4.6.1.3 Local Lunar Terrain Variation

The value $\eta = 0.075$ just defined shall also be interpreted as the present best estimate of the average lunar reflective properties to be viewed by Surveyor radar sensors during approach and descent to the lunar surface. Because of the radar resolution involved, lunar surface structure is expected to cause signal strengths varying both above and below that predicted by the average value. Present best estimates of such variations (including the now small uncertainty in the average) produce the following requirements:

4.6.1.3.1 Upward variations as large as +6 db (required), +10 db (desired) should be allowed for.

4.6.1.3.2 Downward variations as large as -6 db (required), -10 db (desired) should be allowed for.

4.6.1.4 Angle-Dependent Function $F(\theta_i)$

The function $F(\theta_i)$ is defined as:

$$F(\theta_i) = K(\sin\theta_i + \alpha\cos\theta_i)^{-3}$$

where K and α are both wavelength dependent, and $F(\theta_i)$ has a maximum value of (K/α^3) at $\theta_i = 0$ (vertical).

4.6.1.5 Acceptable Parametric Values

Parameter values which are acceptable, without further justification, for purposes required herein are:

Parameter		AMR	RADVS
α		0.36	0.39
(K/α^3)		+10.70 db	+10.15 db
$\eta\ (K/\alpha^3)$:	nominal	- 0.55 db	- 1.10 db
	required min.	- 6.55 db	- 7.10 db
	desired min.	-10.55 db	-11.10 db

These values are based upon the present nominal operating frequencies of 9.3 gcps (AMR), 12.9 gcps (RA), and 13.3 gcps (DVS).

4.6.2 Altitude Marking Radar (AMR)

4.6.2.1 Operating Limits

The AMR shall operate as specified herein for the following conditions:

4.6.2.1.1 Vehicle attitude relative to lunar vertical not to exceed 25°.

4.6.2.1.2 Vehicle velocity relative to lunar surface, in direction of vehicle roll axis, to be between 8,000 and 8,850 ft/sec.

4.6.2.1.3 Vehicle acceleration, in direction of vehicle roll axis, not to exceed 6 ft/sec^2.

4.6.2.2 Altitude Marking Signal

Within the operating limits in 4.6.2.1, an on-off switching signal (the "Altitude Marking Signal") shall be generated, and shall meet the following requirements:

4.6.2.2.1 This signal is used by the Flight Control Subsystem to initiate the terminal descent phase. The time delay between this signal and vernier engine ignition shall be commanded from the ground, and will vary as a function of the approach angle, approach velocity, and midcourse maneuver.

4.6.2.2.2 A correction term will be included in the time delay in 4.6.2.2.1 to compensate for the difference allowed by 4.6.2.2.5 between the desired marking range and the nominal marking range.

4.6.2.2.3 The desired marking range shall be 60 miles (316,800 feet).

4.6.2.2.4 The nominal marking range shall be the predicted average marking range, defined as the arithmetic mean of those two range values at which the cumulative probability of marking is predicted to be 0.001 and 0.999, respectively, as a function of approach angle.

4.6.2.2.5 The nominal marking range as defined in 4.6.2.2.4 shall be within 1.0 mile (5,280 feet) of the desired marking range specified in 4.6.2.2.3, for all approach angles required by 4.6.2.1.

4.6.2.2.6 Random deviations from the nominal marking range as defined in 4.6.2.2.4 shall not exceed ±0.3 mile (1,584 feet)(2) for a vertical approach or ±0.5 mile (2,640 feet)(2) for a 25° approach.

4.6.2.2.7 This signal shall not be produced prior to receipt of the AMR ENABLE command specified in 4.6.2.3.2.

4.6.2.3 AMR Ground Commands

4.6.2.3.1 An AMR POWER ON command shall be transmitted 280 ± 10 seconds(2) prior to the predicted time of the altitude marking signal.

4.6.2.3.2 An AMR ENABLE command shall be transmitted 100 ± 10 seconds (2) prior to the predicted time of the altitude marking signal.

4.6.2.3.3 These requirements are based upon an assumed absolute accuracy of ±10 seconds(2) in the predicted time of the altitude marking signal, based upon ground tracking data and the nominal marking range.

4.6.2.4 False Alarm and Marking Probabilities

The probability of obtaining a false altitude marking signal or of not obtaining a mark signal at all, neglecting equipment failure, shall not exceed 0.002 during the time from AMR ENABLE until a slant range ten miles (52,800 feet) less than the nominal marking range is reached (a maximum time of 127 seconds). For the required equipment reliability of 0.9995 (see 8.3 herein), the resulting required overall probability of proper performance is 0.9975.

4.6.2.5 Alignment Accuracy

As specified in 4.1.3.11 of this specification.

4.6.3 Doppler Velocity Sensor (DVS)

4.6.3.1 Operating Limits

The DVS shall operate as specified herein for the following conditions:

4.6.3.1.1 Vehicle roll (Z) axis relative to lunar vertical not to exceed 45°. Attitude at acquisition not to exceed 25°.

4.6.3.1.2 Maximum required slant range to lunar surface, along vehicle roll axis, not to exceed 50,000 feet.

4.6.3.1.3 Minimum required slant range to lunar surface, along vehicle roll axis, is that value sufficient to insure DVS operation through the RA generation of the second Range Mark Signal specified in 4.6.4.5.

4.6.3.1.4 Vehicle static accelerations along the vehicle roll axis not to exceed:

4.6.3.1.4.1 Before main retro burnout: 380 ft/sec^2

4.6.3.1.4.2 After main retro burnout: 12 ft/sec^2

4.6.3.1.5 The DVS shall operate as required herein in the presence of the separating main retro engine.

4.6.3.1.6 Magnitude of vehicle velocity vector is within the limits:

4.6.3.1.6.1 Before main retro burnout: +100 to +3,000 ft/-sec (descending).

4.6.3.1.6.2 After main retro burnout: +1 to +700 ft/sec (descending) for SC-1 and SC-2; +1 to +850 ft/sec (descending), for SC-3 through SC-7, inclusive.

4.6.3.1.7 The maximum angle between the vehicle roll axis and the vehicle velocity vector at time of DVS acquisition is defined in the following way. Each DVS beam shall be capable of acquiring and tracking (with adequate margin against acquisition of a false signal) a legitimate lunar reflected signal arising from operation within the operating limits herein specified, provided further that the combination of magnitude and direction of the velocity vector does not produce a component of velocity along the DVS beam axis less than the following lower limits:

4.6.3.1.7.1 Before main retro burnout: +100 ft/sec.

4.6.3.1.7.2 After main retro burnout: The greater of +29.6 ft/sec or a straight line through +38 ft/sec at 40,000 feet and +62 ft/sec at 50,000 feet, both ranges being slant ranges to the lunar surface, along the vehicle roll axis.

4.6.3.1.8 Each DVS beam shall also be capable of tracking a legitimate lunar reflected signal, once acquired, producing a velocity component along the axis of that beam as low as +1 ft/sec, provided that signal has adequate power level. Adequate power level shall be no more than required by a reasonable and prudent design capable of meeting the acquisition requirements in 4.6.3.1.7.

4.6.3.2 Outputs

4.6.3.2.1 The DVS shall furnish appropriately scaled DC analog output voltages which respectively represent, consistent with other requirements herein, those three components of vehicle velocity relative to the lunar surface defined by the orthogonal x, y, z axes of the vehicle, where the vehicle roll axis in the normally descending direction is the +z axis.

4.6.3.2.1.1 These Spacecraft velocity outputs shall be derived within the DVS from the separate velocity components normally measured by the several DVS beams.

4.6.3.2.1.2 The separate velocity component indicated by any DVS beam which has not acquired an apparently legitimate lunar reflected signal shall be zero. (Associated error in the spacecraft velocity components thus affected is recognized).

4.6.3.2.2 An on-off switching signal, indicating "Reliable Operation" of the DVS (RODVS), shall be supplied. It shall be in the "on" condition when and only when <u>either</u> all DVS beams are tracking apparently legitimate lunar reflected signals <u>or</u> the CRO signal specified in 4.6.3.2.3 is "on." When this signal exists for the former reason, DVS operation is considered "normal"; when this signal exists for the latter reason, DVS operation is considered to be CRO. (Note that these two conditions are mutually exclusive). This signal is used by the Flight Control Subsystem to switch vehicle attitude control sensing to the DVS outputs.

4.6.3.2.3 An on-off switching signal, indicating "Conditional Reliable Operation" of the DVS (CRO), shall be supplied. It shall be in the "on" condition when and only when at least one but not all DVS beams have acquired and/or are tracking apparently legitimate lunar reflected signals, provided further that:

4.6.3.2.3.1 Its "on" condition shall be inhibited until 4.0 seconds after main retro burnout (see 4.6.3.7 regarding acquisition time).

4.6.3.2.3.2 Its "on" condition shall be inhibited for the remainder of the mission once all DVS beams have been tracking apparently legitimate lunar reflected signals for at least 1.0 second.

4.6.3.2.4 Separate on-off switching signals shall be provided for instrumentation to indicate the search or tracking status of each DVS beam.

4.6.3.2.5 Lunar Reflectivity Signals shall be available as specified in 4.6.3.10, below.

4.6.3.3 Accuracy

In normal DVS operation (as defined in 4.6.3.2.2), and under lunar approach conditions, the accuracies of the analog outputs required by 4.6.3.2.1 shall meet the following requirements:

4.6.3.3.1 Each of the V_x and V_y outputs shall be directly and linearly proportional to its associated true velocity component, within the accuracy specified in 4.6.3.3.3, whenever that true value lies within the region ±500 ft/sec. The V_z output shall be directly and linearly proportional to its associated true velocity component, within the accuracy specified in 4.6.3.3.3, whenever that true value lies within the region: +1 to +700 ft/sec, for SC-1 and SC-2; and +1 to +850 ft/sec, for SC-3 through SC-7 inclusive.

4.6.3.3.2 Whenever any spacecraft true velocity component lies outside the linear region specified in 4.6.3.3.1, but is otherwise within the region specified in 4.6.3.1.6, the corresponding DVS output shall have the correct polarity and a magnitude no less than that corresponding to the maximum linear value.

4.6.3.3.3 In the presence of steady unchanging velocities within the linear region specified in 4.6.3.3.1, the DC or average value of each output, when averaged over any ten-second interval commencing no earlier than 0.5 second after RODVS, shall deviate from the true value by no more than plus or minus the root-sum-square(2) of one ft/sec and two percent of total velocity, for velocities not exceeding +700 ft/sec, or the root-sum-square (2) of one ft/sec and three percent of total velocity, for velocities between +700 and +850 ft/sec.

4.6.3.4 Noise and Ripple

In normal DVS operation (as defined in 4.6.3.2.2), and under lunar approach conditions, and when the vehicle roll axis is within five degrees of alignment with the vehicle total velocity vector, and for true velocities within the linear regions specified in 4.6.3.3.1, and after passing each velocity output required by 4.6.3.2.1 through the proper Spacecraft (S/C) response simulation filter as described below, the instantaneous value over any sixty-second interval in which the true value is essentially constant shall not deviate from the DC value over that interval by more than the following values (which, as defined, have been shown as most nearly representative of three-sigma excursions):

4.6.3.4.1 Each of V_x and V_y (when V_z is 6.5 ft/sec or greater): 0.250 ft/sec, or $(0.083\sqrt{V_z})$ ft/sec, whichever is larger, after a S/C response simulation filter having a transfer function of: $(1)(2.6s + 1)^{-1}(0.11s + 1)^{-2}$. (For test signal inputs which are essentially single-line spectra, rather than real or simulated doppler spectra, the 0.250 ft/sec limit shall apply at all velocity values being simulated.)

4.6.3.4.2 For V_z (without further restriction on its true value): 0.600 ft/sec, or $(0.114\sqrt{V_z})$ ft/sec, whichever is larger, after a S/C response simulation filter having a transfer function of: $(1)(0.08s + 1)^{-2}$. (For test signal inputs which are essentially single-line spectra, rather than real or simulated doppler spectra, the 0.600 ft/sec limit shall apply at all velocity values being simulated.)

4.6.3.4.3 The above shall not preclude the use of S/C response simulation filters having frequency-insensitive gain factors larger than unity, for ease of test instrumentation. In such cases, measured output voltages shall be interpreted by similarly increased electrical scale factors for comparison with the above requirements.

4.6.3.5 Warmup Time

The DVS warmup time shall not exceed thirty (30) seconds, from application of primary power to delivery of the RODVS signal under suitable operational conditions.

4.6.3.6 The DVS shall be capable, after warmup, of all operations required herein until at least 4.5 minutes after application of primary power. Until at least 6.5 minutes after application of primary power, the DVS shall not cause catastrophic failure of any other vehicle equipment because of thermal problems within the DVS.

4.6.3.7 Acquisition

After equipment warmup, the DVS acquisition time after the conditions specified in 4.6.3.1 are simultaneously met shall not exceed 2.0 seconds, with an acquisition probability of at least 0.95, on any DVS beam meeting those conditions. This requirement may be applied separately to operation before main retro burnout and to operation after main retro burnout; i.e., a reacquisition time is allowed after main retro burnout. (Note that CRO is required by 4.6.3.2.3.1 to be inhibited until 4.0 seconds after main retro burnout; this allows two normal acquisition times for normal operation before CRO becomes effective).

4.6.3.8 Response Time

Not to exceed 0.1 second (63%, no overshoot).

4.6.3.9 Alignment Accuracy

The DVS antenna axes shall be aligned relative to the antenna boresight axes within 0.1 degree(1). The overall alignment accuracy shall be as specified in 4...3.12 of this specification.

4.6.3.10 Lunar Reflectivity Signals

Appropriate digital and/or analog signals from each of the DVS receivers shall be made available for instrumentation to provide measures of received signal strengths for lunar reflectivity analyses, and shall meet the following requirements:

4.6.3.10.1 The conversion from the instrumented signal to the appropriate DVS beam signal strength shall be known (after calibration, if necessary) to within ±2 db(6).

4.6.3.10.2 This conversion shall be available over a total dynamic range of input signal strength, per DVS beam, of at least 80 db (desired), 60 db (required).

4.6.4 Radar Altimeter (RA)

4.6.4.1 Operating Limits

The RA shall operate as required herein for the following conditions:

4.6.4.1.1 Vehicle roll axis attitude relative to lunar vertical not to exceed 45°. Attitude at acquisition not to exceed 25°.

4.6.4.1.2 Maximum required slant range to lunar surface, along vehicle roll axis, not to exceed 40,000 feet.

4.6.4.1.3 Minimum required slant range to lunar surface, along vehicle roll axis, is that value sufficient to insure RA operation through the generation of the second Range Mark Signal specified in 4.6.4.5. (For design purposes, RA acquisition may be presumed to have occurred before the slant range has decreased to 2,000 feet before the scale factor change, or to 100 feet after the scale factor change specified in 4.6.4.2.1).

4.6.4.1.4 Notwithstanding any other provisions herein, the RA is not required to operate until after main retro burnout has occurred.

4.6.4.1.5 Vehicle static acceleration along the vehicle roll axis is not to exceed 12 ft/sec(2).

4.6.4.1.6 The RA shall operate as required herein in the presence of the separating main retro engine.

4.6.4.1.7 Vehicle roll axis component of velocity at time of RA acquisition: From +1 ft/sec (descending) to the following upper limits:

4.6.4.1.7.1 For SC-1, -2: A straight line between +560 ft/sec (descending) at 40,000 feet slant range along the vehicle roll axis and +700 ft/sec (descending) at 37,750 feet slant range along the vehicle roll axis, plus a constant upper limit of +700 ft/sec (descending) for slant ranges along the vehicle roll axis less than 37,750 feet.

4.6.4.1.7.2 For SC-3, -4: +850 ft/sec (descending).

4.6.4.1.8 Maximum angle between vehicle roll axis and direction of vehicle total velocity vector is not to exceed 55°.

4.6.4.2 Outputs

4.6.4.2.1 The RA shall furnish an appropriately scaled DC analog output voltage which represents, consistent with other requirements herein, the slant range to the lunar surface, along the vehicle roll axis. To provide the required dynamic range and accuracy, the scale factor of this analog output shall be switched upon generation of the first Range Mark Signal specified in 4.6.4.5. When the RA is operating reliably (as defined in 4.6.4.2.2) and the true range is greater than the maximum required range (or saturation range), the range analog output shall indicate a range no less than the maximum required range.

4.6.4.2.2 An on-off switching signal, indicating "Reliable Operation" of the RA (RORA), shall be supplied. It shall be in the "on" condition when and only when the RA tracker is tracking an apparently legitimate lunar reflected signal and those DVS beams required to obtain proper V_z output from the DVS are also tracking apparently legitimate lunar reflected signals. This signal is used by the Flight Control Subsystem to switch thrust control sensing to the R_z, V_z outputs of the RADVS.

4.6.4.2.3 An on-off switching signal shall be provided for instrumentation to indicate the search or tracking status of the RA beam tracker.

4.6.4.2.4 Range Mark Signals, as specified in 4.6.4.5.

4.6.4.2.5 Lunar Reflectivity Signal(s), as specified in 4.6.4.11.

4.6.4.3 Accuracy

4.6.4.3.1 In reliable operation of the RA as defined in 4.6.4.2.2, and under lunar approach conditions, but assuming a steady and unchanging range, the DC or average value of the output required by 4.6.4.2.1, when averaged over any ten-second interval commencing no earlier than 0.5 second after RORA, shall deviate from the true value (measured along the vehicle roll axis from the boresight plane - that plane, normal to that axis, which contains the two lower boresight balls on the RA antenna) by no more than plus or minus the following amounts:

4.6.4.3.1.1 The root-sum-square(2) of 30 feet and five percent of the true range, for ranges greater than 1,000 feet.

4.6.4.3.1.2 The root-sum-square(2) of four feet and five percent of the true range, for ranges less than 1,000 feet.

4.6.4.3.2 If the RA tracker is tracking an apparently legitimate lunar reflected signal but the other conditions specified in 4.6.4.2.2 are not met (RORA is "off"), the above accuracy may be exceeded to that degree corresponding with the error in the V_z output of the DVS.

4.6.4.4 Noise and Ripple

In reliable operation of the RA (as defined in 4.6.4.2.2), and under lunar approach conditions, and when the vehicle roll axis is within five degrees of alignment with the vehicle total velocity vector, and when the magnitude of the vehicle velocity does not exceed the greater of 10.5 ft/sec or ($4\sqrt{R_s}$) ft/sec (where R_s is the true slant range in feet along the vehicle roll axis), and after passing the output required by 4.6.4.2.1 through a S/C response simulation filter having a transfer function of $(1)(0.08s + 1)^{-2}$, the instantaneous value over any sixty-second interval in which the true value is essentially constant shall not deviate from the DC value over that interval by more than the following values (which, as defined, have been shown as most nearly representative of three-sigma excursions):

4.6.4.4.1 Either 50.0 feet, or $1.264\sqrt{R_s}$ feet, whichever is larger, prior to occurrence of the first range mark signal specified in 4.6.4.5. (For test signal inputs which are essentially single-line spectra, rather than real or simulated spectra, the 50.0 feet limit shall apply at all range/velocity values being simulated.)

4.6.4.4.2 Either 5.0 feet, or $0.424\sqrt{R_s}$ feet, whichever is larger, after occurrence of the first range mark signal specified in 4.6.4.5 but prior to occurrence of the second range mark signal specified in 4.6.4.5. (For test signal inputs which are essentially single-line spectra, rather than real or simulated spectra, the 5.0 feet limit shall apply at all range/velocity values being simulated.)

4.6.4.4.3 The above shall not preclude the use of a S/C response simulation filter having a frequency-insensitive gain factor larger than unity, for ease of test instrumentation. In such cases, measured output voltages shall be interpreted by similarly increased electrical scale factors for comparison with the above requirements.

4.6.4.5 Range Mark Signals

On-off switching signals shall be generated from the analog range output required by 4.6.4.2.1. Each shall be turned "on" at the ranges, under the conditions, and with the accuracies specified below. Once "on", each signal shall remain on as long as the primary power remains within the operating region, or until expiration of the required operating time. These signals shall appear on separate outputs.

4.6.4.5.1 The first range mark signal shall be generated when the analog range output is the voltage equivalent of 1,000 ± 50 feet (2), provided that the vehicle velocity lies within the region 100 ± 25 ft/sec.

4.6.4.5.2 The second range mark signal shall be generated when the true slant range to the lunar surface along the vehicle roll axis is 14 ± 4 feet(2), provided that the vehicle velocity lies within the region 5.0 ± 3.0 ft/sec.

4.6.4.5.3 For vehicle velocities outside the regions specified above, the accuracy of each range mark signal shall be consistent with a reasonable and prudent design capable of meeting the above requirements.

4.6.4.6 Warmup Time

The RA warmup time shall not exceed thirty (30) seconds, from application of primary power to delivery of the RORA signal under suitable operational conditions.

4.6.4.7 Operating Time

The RA shall be capable, after warmup, of all operations required herein until at least 4.5 minutes after application of primary power. Until at least 6.5 minutes after application of primary power, the RA shall not cause catastrophic failure of any other vehicle equipment because of thermal problems within the RA.

4.6.4.8 Acquisition

After equipment warmup, and after main retro burnout, the RA beam acquisition time after the conditions of range, velocity, and attitude specified in 4.6.4.1 are simultaneously met shall not exceed 2.5 seconds for one sweep, or 4.0 seconds for two sweeps, with an acquisition probability of at least 0.95 per sweep.

4.6.4.9 Response Time

Not to exceed 0.1 second (63%, no overshoot).

4.6.4.10 Alignment Accuracy

The antenna axis shall be aligned relative to the antenna boresight axis within 0.1 degree(1). The overall alignment accuracy shall be as specified in 4.1.3.12 of this specification.

4.6.4.11 Lunar Reflectivity Signal

Appropriate digital and/or analog signals from the RA receiver shall be made available for instrumentation to provide a measure of received signal strength for lunar reflectivity analyses. This signal(s) shall conform with the characteristics specified in 4.6.3.10 for the analogous DVS signals.

4.7 Telecommunications

The telecommunications system shall provide a highly reliable two-way communications link between the spacecraft and the DSIF stations during the entire mission. The design of the telecommunications subsystem shall be compatible with the DSIF as defined in the DSIF Requirements Document.

4.7.1 Required Functions

The system shall be capable of performing the following specific functions:

4.7.1.1 Radio reception, demodulation, modulation, and transmission necessary for doppler frequency and angle tracking by the DSIF during the transit phase from injection until just prior to the terminal maneuver.

4.7.1.2 Detection and decoding of radio commands from stations of the DSIF as required to perform the midcourse maneuvers, the terminal maneuvers, scientific instrument control, power management, telemetry data selection, telecommunication subsystem controls and other ground controlled functions necessary to the mission.

4.7.1.3 Processing of all engineering and scientific data signals into a form suitable for transmission via the S/C transmitters.

4.7.1.4 Telemetering of properly processed engineering and scientific data to stations of the DSIF during both transit (including descent and touchdown) and lunar operations phases.

4.7.1.5 Telemetering of television picture information to the DSIF stations.

4.7.1.6 Transmission of a PM signal with phase stability such that the DSIF receiver shall not have a phase error, due to unwanted phase modulation and noise, in the carrier tracking loop in excess of 90.0° zero to peak 3σ.

4.7.2 Transmitters

There shall be two transmitters on the S/C for reliability considerations. Either transmitter shall be capable of being commanded on, one at a time, in either a high power or a low power mode. The power output shall be no less than 39.6 dbm in high power mode and no less than 19.1 dbm in low power mode.

4.7.2.1 Output Connections

Either transmitter can be connected, by command, to any one of the three telecommunications system antennas.

4.7.2.2 Modulation Modes

Each transmitter shall be capable of being modulated in any one of three modes: narrow band phase modulation, narrow band frequency modulation, or wide band frequency modulation.

4.7.2.3 Nominal Center Frequency

The nominal center frequency for all transmitters (except when operating in the transponder mode) shall be same specific frequency such that the actual frequency throughout the operating life shall fall within the band of 2290 mc to 2300 mc.

4.7.2.4 Center Frequency Stability - Long Term

4.7.2.4.1 Narrow Band Phase Modulation Mode

The long term frequency stability shall be such that the transmitter frequency throughout its operating life and over its operating temperature range shall not deviate from the nominal frequency by more than 20 ppm (required) or 5 ppm (desired). The frequency temperature coefficient shall be known prior to launch to within ± 0.1 ppm/°F.

4.7.2.4.2 Narrow Band Frequency Modulation Mode

Same as narrow band PM mode; para 4.7.2.4.1 above.

4.7.2.4.3 Wide Band Frequency Modulation Mode

The long term frequency stability shall be such that the transmitter frequency throughout its operating life and over its operating temperature range shall not deviate from the nominal frequency by more than ± 60 ppm.

4.7.2.5 Center Frequency Stability - Short Term

4.7.2.5.1 Narrow Band Phase Modulation Mode, No Transponder Operation

During zero thrust phases the phase jitter of the unmodulated carrier (excluding antenna effect) shall be such as to cause not more than 36.0° zero to peak (3σ) phase error in a noise free phase coherent receiver, having a loop noise bandwidth of 30 cps.

During retro burning and vernier engine thrusting phase, the phase jitter, as defined above, shall be less than 22.5° zero to peak (3σ) in a noise free, phase coherent receiver, having a loop noise bandwidth of 425 cps.

During landing shocks, resulting from landing conditions specified in 3.11.2, the phase error measured in a noise free, phase coherent receiver, having a loop noise bandwidth of 425 cps shall be less than 30deg zero to peak (3σ).

4.7.2.5.2 Narrow Band Frequency Modulation Mode

Same as 4.7.2.5.1 above plus - when operated in a 20 second on - 40 second off duty cycle, the frequency variation shall not exceed ± 460 cps during an on-off cycle of this operation.

4.7.2.5.3 Wide Band Frequency Modulation Mode

During non thrusting phases the frequency jitter having frequency components in the band of 0 to 220 kc shall be less than 1 kcps zero to peak, and the frequency drift over any 1 second shall be less than ± 46K cps.

During engine thrusting phases, it shall be an objective that these same requirements be met.

4.7.2.6 Modulation Characteristics

4.7.2.6.1 Narrow Band Phase Modulation Mode

This mode is used for the telemetry of the S/C engineering and scientific data whenever carrier phase modulation is required (during transponder operation) or desired. The required characteristics are as follows:

a) Linear dynamic range: ± 2.5 radians

b) Modulation bandwidth: 250 cps to 160 kc at 1 db points

c) Modulation linearity: ± 10% of best straight line characteristics

4.7.2.6.2 Narrow Band Frequency Modulation Mode

This mode is used for the telemetry of the narrow band (Emergency) TV. The required characteristics are as follows:

a) Maximum frequency deviation: ± 5 kc

b) Modulation bandwidth: d-c to 1.2 kc within ± 1.0 db

c) Static frequency deviation linearity: ± 10% of best straight line value over a peak frequency excursion of ± 5 kc

d) Frequency Jitter: The RMS voltage, measured on a discriminator with an inherent output SNR ≥ 50 db, peak-to-peak/RMS and a full scale output of 5 V peak-to-peak shall be ≤150 mv, when demodulating the quiescent carrier in a 1.2 kc bandwidth. This measurement is to be performed at a predetection SNR ≥14 db.

4.7.2.6.3 Wide Band Frequency Modulation Mode

This mode is used for telemetry of the normal TV and for S/C engineering and scientific data whenever carrier frequency modulation is desired. The required characteristics are as follows:

Television Input

a) Maximum frequency deviation: ± 1.65 mc

b) Modulation amplitude response: d-c to 265 kc within ± 1.0 db, with a monotonic decrease beyond 265 kc of at least 6 db per octave (asymptotic)

c) Static frequency deviation linearity: ± 10% or better of best straight line value over a peak frequency excursion of ± 1.65 mc

d) Frequency jitter: The RMS voltage, measured on a discriminator with an inherent output SNR ≥ 50 db peak-to-peak/RMS and a full scale output of 5 V peak-to-peak shall be ≤ 16 mv, when demodulating the quiescent carrier in a 220 kc bandwidth, or ≤ 2.5 mv in a 30 kc postdetection bandwidth. This measurement is to be performed at a predetection SNR ≥ 14 db.

Scientific Input

a) Maximum frequency deviation: ± 1.65 mc

b) Modulation amplitude response: 100 cps to 265 kc, within ± 1 db, within -3 at 10 cps, with a monotonic decrease beyond 265 kc of at least 6 db per octave (asymptotic)

c) Static frequency deviation linearity: ±10% or better of best straight line value over a peak frequency excursion of ± 1.65 mc

d) Frequency jitter: ±1 kc, peak

4.7.3 Receiver

There shall be two receivers on the S/C for reliability considerations. Each receiver shall be operating continuously with its omni-directional antenna.

4.7.3.1 Nominal Center Frequency

When the S/C receiver automatic frequency control or automatic phase control loops are open, the nominal center frequency of the receiver shall be some specific frequency such that the actual operating frequency in all modes of operation shall fall within the band of 2110 mc to 2120 mc.

4.7.3.2 Frequency Stability - Long Term

The long term stability shall be such that the receiver open loop frequency (no APC or AFC due to no input signal to control on) shall not vary throughout its operating range and over its operating temperature from the nominal center frequency, as described above, by more than ±20 ppm. With both receivers having input signals, the open loop frequency of each of the receivers in the S/C at any time shall not be different by more than 40 kc.

4.7.3.3 Frequency Stability - Short Term

4.7.3.3.1 Open Loop Mode

When the receiver is in an open loop frequency control mode (no APC or AFC):

4.7.3.3.2 Closed Loop Mode

4.7.3.4 Frequency Tuning

In the modes of automatic frequency control, AFC and APC, the receiver shall be capable of being tuned over a range of at least ±50 kc and ±70 kc, respectively.

4.7.3.5 Dynamic Range

The receiver shall operate for input signal strengths that vary over at least 40 db exclusive of spacecraft gain uncertainties.

4.7.3.6 Transmitter Signal Rejection

The total isolation between the transmitter and the receiver shall be such that any transmitter signals entering the receiver shall result in no signals falling in the receiver i-f passband having an amplitude greater than 20 db below the receiver noise in a bandwidth of 100 cps.

4.7.4 Antennas

There are three telecommunications antennas aboard the spacecraft. Two are "omni-directional" antennas for command reception and transponder operation, while the third is a high gain antenna capable of providing an effective-radiated-power sufficient for real-time television transmission. In case of antenna failure, an antenna switching function is included to allow use of alternate antennas for transmission.

4.7.4.1 Omni-directional Antennas

The gains and mounting of the two antennas shall be such that the gain in any direction from the spacecraft, considering both antennas, will never be less than -10 db for right hand circular polarization. The gain for the composite pattern of the two antennas for right-hand and left-hand circular polarization will never be less than -7 db except for 2295 mc; at this frequency, in the upper hemisphere, the gain will never be less than -6 db. The gain will be greater than -20 db in any direction from any one antenna over at least 90% of the total 4π steradians. This requirement will be considered as satisfied if the omni-directional antennas have been tested and compared with standard antennas not mounted on the spacecraft.

4.7.4.2 High Gain Antenna

The high gain antenna shall provide a directive beam with a 3 db beamwidth no less than 6.8° x 6.8°. Minimum peak gain shall be 26.5 db.

4.7.4.2.1 <u>Planar Array Boresight Plane</u>

A plane on the planar array defined by boresight tooling markers. This plane is approximately -0.8° from the planar array active face in the same direction as the rf beam depression described in 4.7.4.2.2 below.

4.7.4.2.2 <u>R-f Beam Position</u>

When viewing the planar array with its boresight plane in a vertical plane and the polar axis hingeline perpendicular to the vertical, its beam shall be within ±0.5 degree of the boresight plane perpendicular in a horizontal plane, and depressed 6.5 ±0.5 degree from the horizontal plane. The antenna axis shall be known within ±0.25 degree of the planar array boresight axis at 70°F.

4.7.5 <u>Transponder</u>

The transponder mode is employed during transit and lunar operations to permit two way doppler shift measurements. In this mode, one of the transmitters is driven by one of the receivers through the transponder interconnection circuitry, and the signal transmitted back to earth is coherent with the received signal at a frequency ratio of 240/221. There shall be two transponder interconnection units on the S/C for reliability considerations.

4.7.5.1 <u>Phase Lock Loop Characteristics</u>

4.7.5.1.1 <u>Tracking Range</u>: ±70 kc at threshold+

4.7.5.1.2 <u>Tracking Rate</u>

The phase locked loop shall be able to track a threshold signal, have a rate of change of frequency up to 1 kc/sec^2 lasting over a duration of one minute (subject to the tracking range limitation given in 4.7.5.1.1 above).

4.7.5.1.3 <u>Acquisition</u>

The loop shall be able to acquire under either of the following conditions:

(a) When a threshold+ signal is sweeping through the bandpass at a rate of 1 kc/sec^2.

(b) When a non-sweeping threshold+ signal is within 500 cps of the open loop rest frequency of the receiver.

+Up-link Threshold is defined as -114 DBM for total signal power (carrier and subcarrier) when the subcarrier/carrier modulation index is 1.6, measured in a predetection equivalent noise bandwidth of 13 kc.

4.7.5.1.4 Loss of Lock

With threshold signal* levels, the S/C transponder shall not lose lock more than once per 10 minutes.

4.7.5.2 Transponder Phase Stability Characteristics

While in transponder operation, the phase stability of the carrier shall have the following characteristics:

During zero thrust phases, the phase jitter of the unmodulated carrier (excluding antenna effect) shall be such as to cause not more than 36.0° zero to peak (3σ) phase error in a noise free phase coherent receiver, having a loop noise bandwidth of 30 cps.

During retro burning and vernier engine thrusting phases, the phase jitter, as defined above, shall be less than 22.5° zero to peak (3σ) in a noise free, phase coherent receiver, having a loop noise bandwidth of 425 cps.

During landing shocks, resulting from landing conditions specified in 3.11.2, the phase error measured in a noise free, phase coherent receiver, having a loop noise bandwidth of 425 cps shall be less than 30 degrees zero to peak (3σ).

4.7.6 Signal Processing

The signal processing system shall be capable of converting all engineering and scientific data signals into a form suitable for modulation of the spacecraft transmitter in accordance with the data modes specified in section 4.7.7. To maintain flexibility in the instrument payload, as much processing as possible shall be associated with individual instruments and subsystems. Major functions of the central signal processor associated with the spacecraft are: analog to digital conversion, commutating time and frequency division multiplexing, amplification, weighting, and summation of signals for phase and frequency modulation of the transmitters. The signal processing system shall be compatible with the wide and narrow transmission bandwidths of the data link system.

4.7.6.1 Data Accuracy

The overall 3σ accuracy of the various types of telemetered data including the sensor, the S/C signal processing, the S/C to ground communication link when operating at threshold* signal levels, shall be as given below. These requirements are consistent with the following assumptions for specification purposes:

a) Noiseless demodulation

b) Zero error readout devices

c) Efforts of bit errors removed by parity checking

d) Use of sensor calibration, if necessary

		Accuracy	Required Bandwidth (cps)
4.7.6.1.1	High Accuracy, PCM Temperature Data:	1°C for temperatures below 100°C	0.1
4.7.6.1.2	Basic Accuracy, PCM, Temperature Data:	2°C (desired) 4°C (required)	0.1
4.7.6.1.3	PCM Power Supply Current Data:	1 percent	0.5
4.7.6.1.4	PCM Potentiometer Readout Data:		
	a) ASPP Position Signals	0.9 degree	0.5
	b) Other potentiometer data	3%	0.5
4.7.6.1.5	Flight Control Data		
	a) Gyro Error (Pitch and Yaw)	5% Desired 10% Required	3.0
	b) Precession Command	5% Desired 10% Required	0.5
	c) Primary Sun Sensor Error (Pitch and Yaw)	5% Desired 10% Required	0.4
	d) Canopus Error	5% Desired 10% Required	0.5
	e) Star Intensity	3% Desired 6% Required	0.3
	f) Secondary Sun Sensor Center Cell Signal Indicating Angle of S/C - Sun Vector to Sensor Boresight Axis	0.75 degree	0.5
	g) Acceleration Error	10% Desired 20% Required	5.0

		Accuracy	Required Bandwidth (cps)
h)	Retro Accelerometer	5% Desired 10% Required	5.0
i)	Roll Actuator	10% Desired 20% Required	5.0 0.4
j)	Radar Altimeter Range Range < 1000 ft Range > 1000 ft	 R.S.S.(5 and .055R) ft R.S.S.(36 and .055R) ft	0.4
k)	RADVS, V_x and V_y	R.S.S.(1.2 and .025V_t) fps	0.5
l)	RADVS, V_z	R.S.S.(1 and .02V_t) fps	1.0
m)	RADVS Amplitude	6 db	
n)	Vernier Engine Thrust Commands	5% Desired 10% Required	5.0
o)	Vernier Engine Strain Gage	5% Desired 10% Required	5.0

4.7.6.1.6 Data Link Data

a)	Static Phase Error	10%	0.5
b)	Receiver AFC	10% Desired	0.5
c)	Receiver AGC	2 db	0.5
d)	Omni Transmitted Power	2 db	0.5

4.7.6.1.7 Altitude Marking Radar Data

a)	AMR Magnetron Current	5% Desired 10% Required	0.5
b)	AMR AGC, after AGC is effective	2 db Desired	1.0
c)	AMR Late Gate Signal, indicate return signal power before AGC is effective	2 db Desired	2.0

4.7.6.1.8 Frequency Modulation Analog Data — 5% Desired, 10% Required

4.7.6.2 Bit Error Probability

The recovered PCM data at minimum margin threshold SNR of 11 db (RMS/RMS) shall have a bit error probability of no greater than 3×10^{-3}. This gives a word error probability of approximately 1 in 30. During Coast Phase II, a word error rate of no greater than 1 in 6 shall be allowed. This reduction in word error rate implies that the threshold SNR is less than 11 db during this phase.

4.7.7 <u>Bandwidth Requirements</u>

4.7.7.1 <u>Transit Phase</u>

Given in the table below are the information bandwidths or data rates required under the various conditions of the transit phase to telemeter the necessary engineering data.

Phase	Data Mode	Equip. Mode		Range (km)	Bandwidth	
		Xmtr Power	Ant		Bit Rate (bps)	Info. (kc)
Coast	1 Channel - PCM	Low	Omni	406,700	17-3/16	-
Midcourse	1 Channel - PCM	High	Omni	230,000	4400	-
Terminal descent - retro and vernier	1 Channel - PCM	High	Omni	406,700	1100	-
Terminal descent - retro	1 Channel - PCM plus multi-analog channels.	High	Array	406,700		220

4.7.7.2 Lunar Phase

Given in the table below are the information bandwidths or data rates required under the various phases of lunar operation to telemeter the necessary scientific and engineering data.

Phase	Data Mode	Equipment Mode		Bit Rate (bps)	Bandwidth	
		XMTR Power	Antenna		Information	IF
Normal TV	Single channel - FM	High	Array		220 Kcps	3.3 mc
Emergency TV	Single channel - FM	High Low	Omni Array		1.2 Kcps 1.2 Kcps	10.5 kc 10.5 kc
Low Data Rate - Engineering	Single channel - FM/PM	Low Low High High	Omni Array Omni Array	17-3/16 NRZ 1100 NRZ 550 NRZ 4400 NRZ		
	Single channel - FM	Low High High	Array Omni Array	--- --- 4400 NRZ		
Low Data Rate-Scientific (SC-5, 6, 7)	Multiple Channel - FM/PM:	Low	Array			
	Engineering Commutator			550 NRZ		
	Alpha Scattering			2200 NRZ		
	Alpha Count Proton Count			550 NRZ		

4.7.8 Command Channel Requirements

4.7.8.1 Bit Error Probability

The spacecraft recovered command data at threshold shall have a bit error probability of no greater than 2×10^{-5}. This gives a command reject probability of 1 reject/2000 commands or approximately 1 reject every 30 minutes during a TV survey.

4.8 Mechanisms

For purposes of establishing the required mechanism positioning ranges it shall be assumed that the landing evelope will be no less than 15 degrees from the limb and the maximum lunar slope of 15 degrees shall be long relative to the spacecraft height.

4.8.1 Planar Array

Orientation of the planar array with respect to the S/C coordination system is determined by three variable angles: (1) roll, (2) elevation, and (3) polar (see Figure 4-2). Roll, elevation, and polar angles may be changed by radio command.

4.8.1.1 Positioning Range by Radio Command

4.8.1.1.1 Roll Angle

Zero reference position for the roll angle shall be that position where the elevation axis is parallel to the spacecraft x (pitch axis) and the active face of the planar array is facing along the minus x (pitch) axis. The planar array shall rotate ±190°.about the zero roll position. Positive roll is as defined in Figure 4-4. There shall be provided the capability to control the position of the planar array to within ¼ degree of any commanded position.

4.8.1.1.2 Elevation Angle

Zero reference position for the elevation angle shall be that position in which the polar axis is perpendicular to the roll axis. The planar array shall rotate about the elevation axis ±90° from the zero reference position. Positive elevation angle is as defined in Figure 4-4.

4.8.1.1.3 Polar Angle

Zero reference position for the polar angle shall be that position in which the planar array's boresight plane is perpendicular to the xy plane. The planar array shall rotate from +96 degrees to +6 degrees from the zero reference point. (See 4.7.4.2.1 for definition of planar array boresight plane.) Positive polar angle is as defined in Figure 4-4.

4.8.1.2 Axes Alignment

All adjoining axes shall be perpendicular to each other within ±0.50 degree(2).

4.8.1.3 Positioning Accuracy

4.8.1.3.1 Positioning Mechanism

The positioning mechanisms in roll and elevatior axes shall be capable of moving in nominal steps of no less than 1/8 degree and no more than 1/4 degree. The positioning mechanism in polar axis shall be capable of moving in nominal steps of no less than 1/16 degree and no more than 1/4 degree. There shall be provided the capability to control the position of the planar array to within 1/4 degree of any commanded position.

From the start of the pre-retro maneuver through vernier descent, when not commanded to rotate, the planar array shall not rotate by more than 1 degree about the spacecraft Y axis (ASPP polar axis) nor more than 0.4 degree about the spacecraft Z axis (ASPP roll axis) as a result of gear and drive backlash and all other causes.

4.8.1.3.2 Position Instrumentation

Instrumentation shall be capable of indicating the angular rotation about each axis with respect to the zero reference point (4.8.1.1) within 1.5 degree(2).

4.8.1.3.3 Adjustment Range, Landing

The planar array shall be positioned, in increments of 0.25 degree or less, such that the polar axis landing angle is within the range of zero to +70.5 degrees relative to the zero position (see 4.8.1.1.1). This

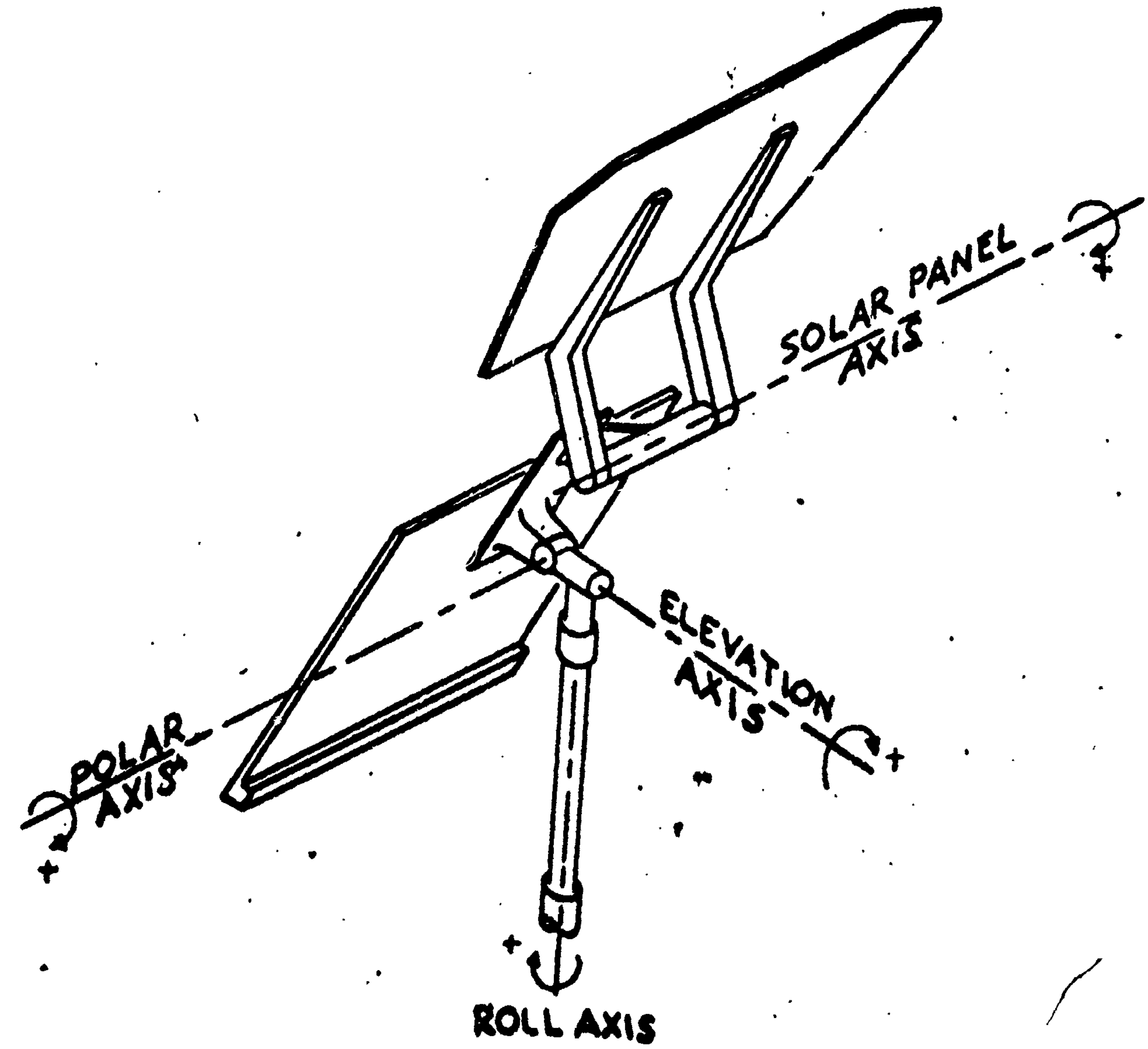


FIGURE 4-4 SOLAR PANEL AND PLANAR ARRAY AXES	224510	
SURVEYOR LABORATORY HUGHES AIRCRAFT COMPANY SPACE SYSTEMS DIVISION	H	SC
	102 177	

is equivalent to a range of "planar array antenna axis 'to' S/C z axis" angles between 60 degrees and 26 degrees. The shift in combined center of gravity produced by positioning the planar array within this range shall be as defined in paragraph 4.1.3.3.1.5.

4.8.1.3.4 Roll Axis Alignment

The planar array actuating mechanism roll axis shall be parallel to the S/C z (roll) axis within 1.0 (1) degree. The mechanism elevation axis shall be parallel to the S/C x (pitch) axis within 1.0(1) degree when the roll axis is in the transit lock position. Additional allowances for the effect of shock are defined in paragraphs 4.8.1.3.5 and 4.8.2.3.5. When the roll and elevation axes are locked in transit position, the planar array boresight plane (see 4.7.4.2.1) shall be perpendicular to the S/C x (pitch) axis within 1 degree(1).

4.8.1.3.5 Landing Load Deflections

Following all S/C lunar landings resulting in not more than 40g lateral loads at the antenna, the planar array antenna axis shall be within ±0.4 degree(2) of its position prior to the landing. This deflection shall be made up of no more than 0.2 degree(2) spaceframe deformation and no more than 0.2 degree(2) mast and mechanism deformations.

4.8.1.4 Positioning Rate

It shall be possible to drive the planar array about its roll axis at any time during transit except when this axis is locked at a rate of 15° per minute. It shall be possible to drive the planar array about its polar axis at any time during transit at a rate of 7.5° per minute. The planar array shall not be rotated about its elevation axis during transit.

At launch, the roll angle may be up to 60 degrees away from the zero reference position as defined in 4.8.1.1.1.

4.8.1.6 Transit Position

During transit from after initial positioning until pre-retro maneuver, all 3 axes shall be in their zero positions (as defined in 4.8.1.1).

4.8.1.7 Lunar Phase Positioning

For SC-2 through SC-4, it shall be possible to position any of the antenna/solar panel positioner drive motors with a duty factor of at least 25 percent with the maximum allowable active period of the cycle of at least the time determined as a function of temperature from the curves of Figure 4-4-1. For SC-5 through SC-7, it shall be possible to position any of the antenna/solar panel positioner drive motors with a duty factor of at least 40 percent and with the capability of the active portion of the cycle to be at least 15 minutes.

The A/SPP shall be capable of at least 97 percent response to commands over the entire range of movement. This requirement applies to all conditions of S/C power and over the entire range of operating environments.

4.8.1.8 Operating Life

The minimum operating life after launch of each of the 3 drive axes shall be:

	Required Steps in each Direction	Total
Roll Axis -	16,000	[illegible],000
Polar Axis -	10,000	20,000
Elevation Axis -	10,000	20,000

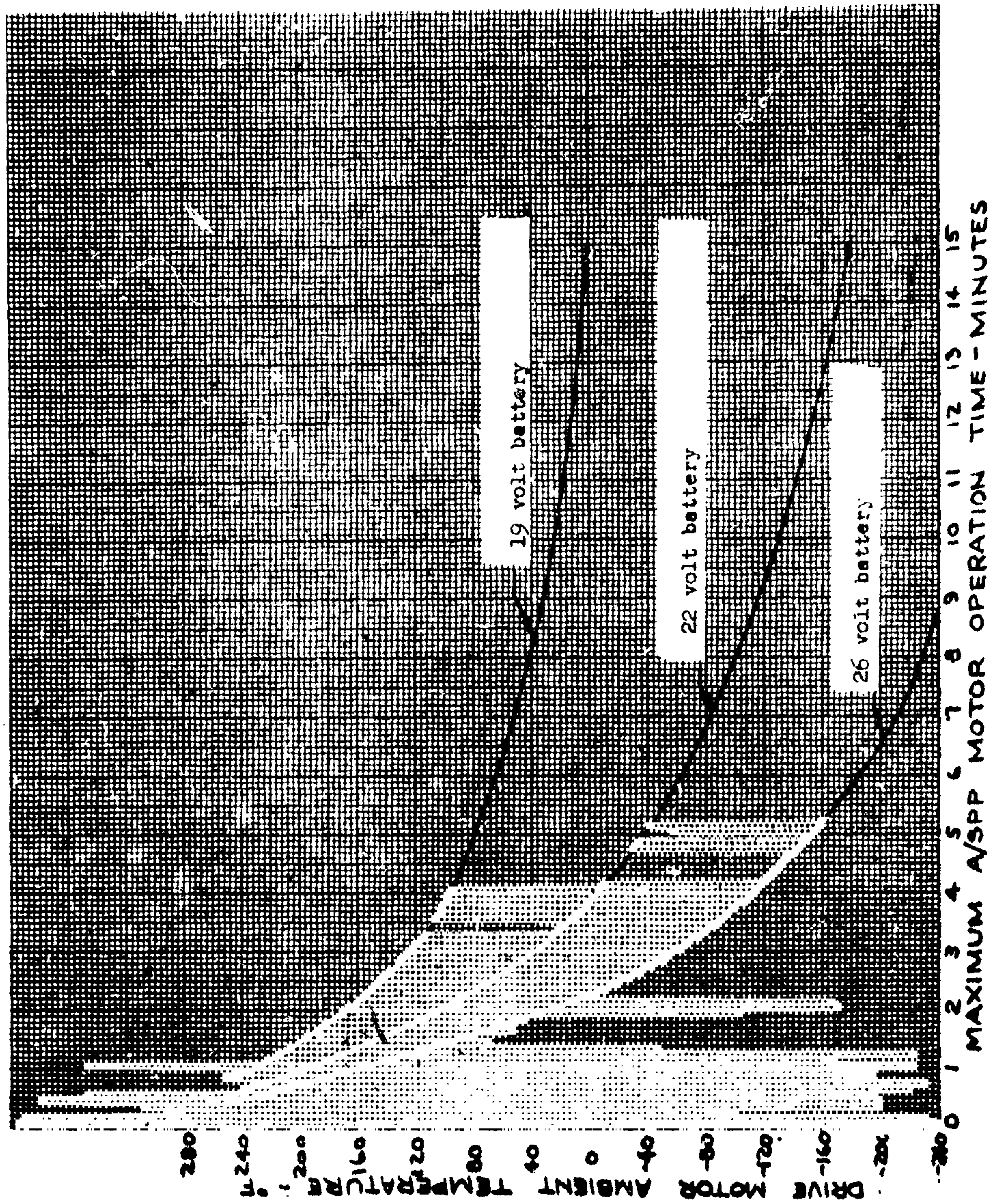


Figure 4.4.1 ASPP Continuous Stepping Constraint

4.8.2 Solar Panel

Orientation of the solar panel with respect to the planar array is determined by a variable angle about the solar panel axis. This angle can be changed by radio command or by automatic-deployment-logic circuitry aboard the spacecraft. Although positioning accuracy of the solar panel is not critical, positioning of the secondary sun sensor mounted on the solar panel is critical. For this reason, the solar panel plane shall be defined as the secondary sun sensor mounting surface.

4.8.2.1 Positioning Range

The solar panel shall be capable of being positioned as a function of polar angle to the solar panel cycles shown in Figure 4-5. The solar panel angle is defined as the angle measured from the planar array boresight plane to the solar panel plane defined in 4.8.2. Positive solar panel angle is as defined in Figure 4-4.

4.8.2.2 Positioning Axis

A plane perpendicular to the polar axis shall be perpendicular to the solar panel axis within 0.5 degree (2).

4.8.2.3 Positioning Accuracy

4.8.2.3.1 Positioning Mechanism

The solar panel axis positioning mechanism shall be capable of moving in nominal steps of no less than 1/8 degree or no more than ¼ degree. There shall be provided a capability to control the position of the solar panel to within 3/4 degree of any commanded position.

4.8.2.3.2 Position Instrumentation

Instrumentation shall be capable of indicating the angle of rotation about the solar panel axis within 2.0 degrees (2).

4.8.2.3.3 Solar Panel Plane

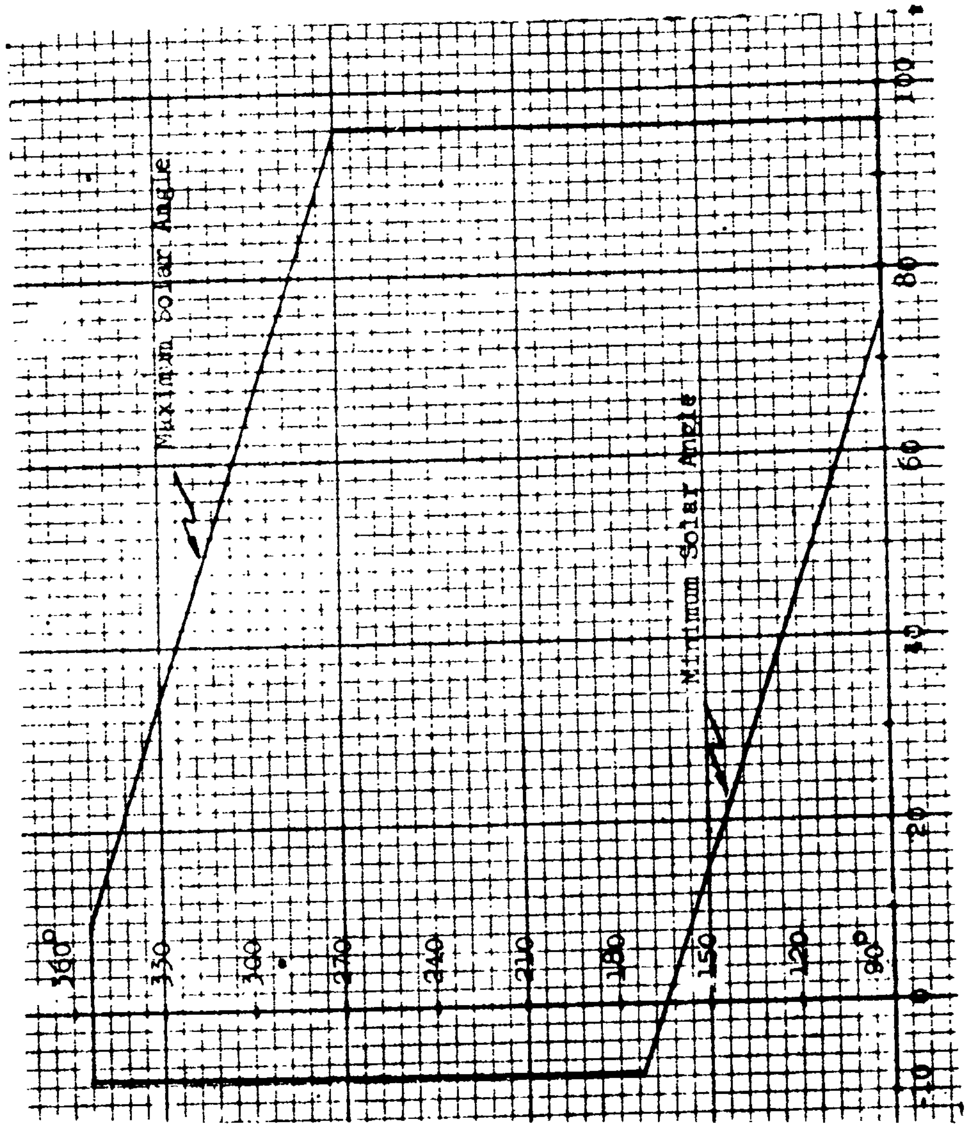
The solar panel plane shall be parallel to the solar panel axis within ±0.50 degree(2).

4.8.2.3.4 Solar Panel Active Surface

The solar panel active surface shall be parallel to the solar panel plane within 2.0 degrees(2).

SOLAR ANGLE POSITION RANGE AS A FUNCTION OF POLAR ANGLE

Figure 4-5



POLAR ANGLE, DEGREES

SOLAR ANGLE (DEGREES)

TITLE	NUMBER
FIGURE 4-5 SOLAR ANGLE POSITION RANGE AS A FUNCTION OF POLAR ANGLE	224510
SURVEYOR LABORATORY HUGHES AIRCRAFT COMPANY SPACE SYSTEMS DIVISION	REVISION H / MODEL SC PAGE 107 OF 177 PAGES

4.8.2.3.5 Landing Load Deflections

Following all S/C lunar landings resulting in not more than 25g lateral loads at the solar panel, the solar panel plane shall be parallel within ±0.50 degree(2) of its position prior to the landing. This deformation shall be made up of ±0.2 degree(2) spaceframe deformation and 0.3 degree(2) mast and mechanism deformation with respect to the roll axis.

4.8.2.4 Positioning Rate and Life

4.8.2.4.1 Transit Positioning Rate – During the transit phase, it shall be possible to drive the solar panel about the solar axis at a rate of 15 degrees per minute with a 100 percent response to commands.

4.8.2.4.2 Lunar Positioning Rate – See Paragraph 4.8.1.7.

4.8.2.4.3 Operating Life – The minimum operating life, after launch, of the solar panel axis drive shall be 16,000 steps in each direction or a total of 32,000 steps.

4.8.2.5 Transit Position

During transit from after initial positioning until pre-retro maneuver, the nominal solar panel angle (as defined by paragraph 4.8.2.1) shall be 270 ± 1.0 degrees.

4.8.2.6 Solar Panel Mounts

In providing for the solar panel mount, the total mechanism (excluding the mast mounting and the spaceframe) shall contribute less than 1.0 degree(1) to the secondary sun sensor alignment error during transit (sec 4.1.3.15).

4.8.2.7 Secondary Sun Sensor Mounts

In providing for mounting of the secondary sun sensor, the solar panel shall contribute less than 0.5 degree(1) to the secondary sun sensor alignment error (see 4.1.3.15).

4.8.3 Gimbal Locks

The ability to drive any one of the four axes shall be independent of the locking mechanisms on the other axes.

4.9 Electrical Power System

4.9.1 Solar Power Conversion

4.9.1.1 Transit Phase

For SC-2 through SC-4, during the transit coast phases, when the solar panel is oriented perpendicular to the spacecraft-sun line, the mean solar panel output shall be at least 89 watts (the predicted solar cell temperature at such time is +122°F ±10°F). Assuming a mean solar illumination intensity of 140 milliwatts/cm^2 and a solar panel power output varying linearly with illumination intensity, the following equation shall describe the possible variations in the transit power value band: Power, transit = 1 ±.035 89 ± 5 (±3) watts, where ±.035 represents the seasonal variation in solar intensity. For SC-5 and up, the redesigned solar panel shall supply 81 watts at a temperature (transit) of 143°F. For lunar operation the redesigned solar panel shall have an open circuit voltage of 28 volts (minimum) and a voltage of 22 volts (minimum) at the maximum power transfer point at a temperature of 250°F.

4.9.1.2 Lunar Phase

During the first lunar day operation, under conditions of an equatorial landing and solar panel orientation perpendicular to the spacecraft-sun line, the following schedule of the mean power output shall be available at the output of the solar panel:

Solar angle from the zenith (degrees)	Mean solar panel output (watts)	Maximum solar panel temperature °F*
0 (noon)	63.1	236 ± 10
±45	67.4	214 ± 10
±60	68.6	208 ± 10
±85	78.9	156 ± 10
±90 (Start of sunset)	80.1	148 ± 10

4.9.2 Energy Storage Capability

4.9.2.1 Auxiliary Battery Capability

4.9.2.1.1 Capacity

An auxiliary battery shall be provided with a minimum recoverable capacity of 900 watt-hours under the load conditions existing at the terminal phase of transit (maximum of 70 amperes) assuming a stand time of 15 days and a temperature of 80°F. A capacity of at least 750 watt hours is required assuming a stand time of 22 days under the above conditions. Maximum time from activation to touchdown shall be 22 days.

4.9.2.1.01 The Auxiliary Battery Capability shall be deleted from SC-5,6 and 7.

4.9.2.1.2 Battery Voltage

Under the above conditions the auxiliary battery shall have a capability to supply a minimum of 16.5 volts at the battery terminals.

4.9.2.2 Main Battery

4.9.2.2.1 Capacity

The battery shall provide energy in accordance with the following table, assuming a float charge to 26.9 volts and 250 milliamps at 70°F or greater:

	Minimum Recoverable Energy watt-hours		Temperature °F	Constant Load amperes
	Nominal	Tolerance(2		
1. Initial charge (charge at launch) or second charge (full recharge during lunar day) available for lunar night	3450	±200	70°F	2
	3500	±200	70°F	1
	3120	±200	50°F	1
	2630	±200	30°F	1
	2300	±200	10°F	1

4.9.2.2.2 Load

The battery shall provide a minimum of 17.5 volts at the terminals under the peak load at 70 amperes with a temperature ranging between 70° and 125°F.

4.9.2.2.3 Lunar Day Temperatures

The battery shall be capable of withstanding compartment thermal tray temperatures from 0°F to +125°F, without permanent degradation of performance.

4.9.3 Power Control System

4.9.3.1 Optimum Charge Regulator

In order to provide operational flexibility to adjust the level of power dissipated in compartment A, or the power generated for supplying the unregulated bus, the power system shall be capable of 3 modes of ground control. In addition, a capability shall be provided to charge the battery when the battery terminal voltage is zero volts (not shorted), independent of command capability. This capability shall extend over the open-circuit solar panel voltage range of from 32 volts to 60 volts.

a) Optimum charge regulator on - for optimum transfer of electrical power from the solar panel to the battery. The OCR in the normal mode shall, in the transit phase of the mission, supply sufficient energy for soft landing and for the completion of the post-landing operations specified in paragraph 3.13.5.1.

b) Optimum charge regulator by pass mode - for transfer of electrical power from the solar panel to the battery at a lower dissipation of power in compartment A.

c) Optimum charge regulator off - for the removal of the solar panel from the system and for the elimination of heat dissipation in the compartment by the elements of the battery charging circuitry.

d) For SC-5 and up, the OCR will be replaced by a capability of directly connecting the redesigned solar panel to the preregulated or unregulated bus.

4.9.3.2 Power Regulation

Regulated power shall be provided consistent with the requirements of the basic bus and engineering or scientific payload. In order to provide the operational flexibility to control the level of power dissipated in compartment B, the capability shall be provided, for use during the lunar phase, to turn off the regulator such that only the command receivers and decoder selector operates from the unregulated bus.

4.9.3.3 Battery Switching Logic

The utilization of the auxiliary battery shall be primarily as a redundant battery in the event of main battery failure and secondarily as an energy source to maximize the main battery state of charge at touchdown. Switching logic shall therefore be provided to apply either the main battery alone or both batteries simultaneously to the unregulated bus. Simultaneous operation shall be with the main battery either direct or through a diode and with the auxiliary directly connected.

4.9.3.4 Power System Instrumentation

A capability shall be provided to measure the regulated and unregulated loads (pulsed loads excepted) to a maximum uncertainty of 2% (4). A three level calibration of the current shunt voltages shall be provided (4) for use during operations which will reduce the above uncertainty to 1% (4) required, (0.5% (4) desired), neglecting load fluctuations occurring during the interval between the calibration and load sampling.

4.10 Thermal Control

Spacecraft temperature control shall be provided to maintain temperatures consistant with the equipment operating requirements during the transit and lunar phases of the mission.

4.10.1 Spacecraft Attitude

4.10.1.1 Transit Phase

Normally the spacecraft will be oriented such that its +Z axis is pointing toward the Sun. However, the thermal control shall be such as to permit the following non-normal conditions to exist without resulting in any permanent damage to the spacecraft or in any violation of the operating temperature requirements of the subsystems necessary to accomplish a successful landing.

"4.10.1.1.1 Initial sun acquisition must take place not later than one hour after launch. The spacecraft may therefore be in the earth's shadow for up to 42 minutes after launch or at random attitudes to the sun for up to one hour after launch."

"4.10.1.1.2 The spacecraft may be in the earth's or moon's shadow or at a random attitude to the sun during transit for a period of one half hour provided that it has not been in shadow or at random attitude to the sun for the preceding five hours."

4.10.1.2 Lunar Phase

For purposes of thermal control, the spacecraft roll orientation relative to the Sun shall be assumed to be random. In addition, the angle between the spacecraft -Z axis and the vertical shall be assumed to be nominally 7 degrees with a maximum of 15 degrees.

4.10.2 Thermally Controlled Compartments

Two thermally controlled compartments provide the necessary temperature control for the operation of electronics during the transit and lunar mission phases. The temperature control capability utilized includes the transient as well as steady-state response of the compartment thermal trays to the heat generated internally by the electronic units. The above performance parameters are influenced by two additional parameters. These are: 1) Thermal heat capacity and 2) Electrical power dissipated internally as heat. It is the purpose of this section and subsequent subsections to specify the basic compartment thermal control capability only to the thermal interface which exists between the thermal trays and the electronic units mounted to the trays. The thermal design requirements on the individual electronic units and the operationally controlled electrical power dissipation requirements are defined in subsequent sections.

4.10.2.1 Instrumentation

One flight sensor shall be mounted on each compartment upper and lower thermal tray for the purpose of providing a valid indication of the thermal status of each thermal tray which acts as a heat sink for the attached electronics units.

4.10.2.2 Thermal Capacity

4.10.2.2.1 For purposes of compartment thermal design, the thermal heat capacity of compartment A shall be assumed to be________.

4.10.2.2.2 For purposes of compartment thermal design, the thermal heat capacity of compartment B shall be assumed to be________.

4.10.2.3 Compartment A and B Thermal Parameters

4.10.2.3.1 Transit Phase

4.10.2.3.1.1 Steady-State Performance

The maximum steady-state temperatures of the compartment A and B thermal trays as indicated by their respective sensors, under normal coast phase environmental conditions, as a function of internal electrical power dissipated as heat shall be as described in Figure 4-6. The temperature gradients relative to the sensors shall be no greater than indicated in Figure 4-6.

4.10.2.2.1.2 Transient Performance

The maximum rate of change of the compartment A and B thermal trays as indicated by their respective sensors, under normal coast phase environmental conditions, as a function of internal electrical power dissipated as heat shall be as described in Figure 4-7. The temperature gradients relative to the sensors shall be no greater than indicated in Figure 4-7.

4.10.2.3. Lunar Phase, Lunar Day

4.10.2.3.1 Steady-State Performance

The maximum steady-state temperatures of the compartment A and B thermal trays as indicated by their respective sensors as a function of internal electrical power dissipated as heat shall be as described in Figure 4-8. The temperature gradients relative to the sensors shall be no greater than indicated in Figure 4-8.

4.10.2.3.2 Transient Performance

The maximum rate of change of the compartment A and B thermal trays as indicated by their respective sensors as a function of internal electrical power dissipated as heat shall be as described in Figure 4-9. The temperature gradients relative to the sensors shall be no greater than indicated in Figure 4-9.

4.10.2.3.3 Lunar Night Compartment Heat Losses

4.10.2.3.3.1 The thermally controlled compartments A and B shall be thermostatically controlled. At compartment temperatures of 30°F and 10°F, respectively, the lunar night heat losses shall be equal to or less than the following:

Maximum Heat Loss, Watts	
Compartment A	Compartment B
10.0	7.0

4.10.3 Compartment Electronics

Thermal design requirements are placed on the compartment electronics units in order to provide the necessary operational capability consistent within the established thermal interface at the thermal tray. The purpose of this section is to specify the thermal design requirements which relate to the electronics units only.

4.10.3.1 Instrumentation

A flight sensor shall be mounted in a critical area on each major heat dissipating unit for the purpose of providing a valid indication of the thermal status of each such unit.

4.10.3.2 Thermal Parameters

Any electronics unit internal to compartment A and B shall be capable of continuous operation when the thermal tray at the unit mounting is maintained equal to or less than +125°F.

4.10.3.3 Non-Operating Requirements

The electronics internal to the compartments A and B, with the exception of the main battery, shall be capable of withstanding without permanent degradation a thermal tray temperature of from -65°F to +150°F.

4.10.4 Compartment A and B Internal Electrical Power Dissipation

The electrical power dissipated as heat in compartment A and B by the internally mounted electronics units shall be operationally controlled such that the heat dissipated is consistant with the compartment parameters specified in sections 4.10.2 and 4.10.3. The average electrical power dissipated as heat shall be limited in accordance with the following table.

Mode	Time Interval	Maximum Average Electrical Power Dissipated at Heat, Watts	
		Compartment A	Compartment B

5.0 ENGINEERING PAYLOAD

5.1 Engineering Mission

The primary objective of the Surveyor engineering mission as stated in the JPL SRS is "to demonstrate successful transit and soft lunar landing, and to obtain engineering data relative to the performance of the S/C in transit and in lunar environments." The payload for engineering missions consists of the auxiliary battery and control, the survey television camera, the auxiliary engineering signal processor and supplementary engineering sensors.

For SC-[illegible], [illegible] and 7, the auxiliary battery and control shall be deleted and an alpha scattering instrument shall be added.

5.2 Survey Television

The survey television system is considered here as an integrated system consisting of the camera as mounted in the spaceframe, the television auxiliary, [illegible], spacecraft transmitter, the spacecraft antenna, the DSIF demodulator, and the DSIF receiver. Except where specifically noted, the system requirements herein pertain to the above system. Primary demodulation and photorecording is accomplished by the JPL TV Ground Data Handling System.

The system allows observation of portions of the spacecraft and its operation, and is capable of employing color or polarizing filters. In the event of failure of the high gain antenna or high power transmitter, the system will provide an "emergency" mode at reduced scan rate and lower picture quality compatible with the transmission bandwidth under these conditions.

In the event of a night landing the system will have the capability of providing indefinite time exposures to obtain a few pictures of somewhat lower quality from earthlight.

5.2.1 Extent of Information

The television system shall be designed as far as possible to meet the following goals for the minimum design mission for daylight landings.

a) Subject to the thermal constraints of 4.10 and 5.2.2.20, one complete wide angle survey and one complete in-focus, narrow angle mapping sequence with camera 3 during the first Goldstone visibility period (i.e., the period from vehicle touchdown to the passing of vehicle control to Canberra for the first time), will be made.

b) At least three further complete mapping sequences are to be performed at approximate sun declination intervals of 30 degrees each before local terminator.

All of Section 5.0 has been revised since Revision D.

c) At least two complete surveys with the three color filters will be made.

d) Pictures will be made as required of portions of the spacecraft and its associated mechanisms to verify proper operation, monitor subsystem mechanical failures for diagnostic purposes, obtain relevant information regarding properties of dust possibly adhering to the spacecraft, and to determine the attitude of the camera in S/C coordinates after landing.

e) Pictures of stars and planets will made to provide accurate information regarding spacecraft attitude. Although only two such fixes are actually required to fix attitude, a greater number is desirable for camera boresighting, and magnification calibration.

5.2.2 System Performance

5.2.2.1 Azimuth Coverage

The camera shall be capable of 360 degrees azimuth viewing.

5.2.2.2 Elevation Coverage

The camera shall be capable of viewing from no less than 20 degrees above to no less than 45 degrees below the spacecraft XY plane.

5.2.2.3 Unobstructed Coverage

Camera mounting shall be such as to maximize unobstructed (by the spacecraft) azimuth and elevation coverage consistent with other constraints.

5.2.2.4 Field of View

Two modes ("narrow angle" and "wide angle") shall be provided. For infinite object distance the nominal field of view shall be 6.4 x 6.4 degrees in narrow angle and 25.2 x 25.2 degrees in wide angle. Spacecraft 5, 6, and 7 may have up to 15 percent loss of raster angular coverage due to overscanning.

5.2.2.5 Normal Mode System Horizontal Relative Response (Resolution)

After appropriate filtering of the demodulated video signal, the square wave relative response $V_H(N)$ shall be equal to or exceed that shown in Figure 5-2 under laboratory ambient conditions at a single measured value of signal-to-noise ratio as defined in 5.2.2.9. Up to 4 db degradation at the extremes of the operating temperature range is acceptable.

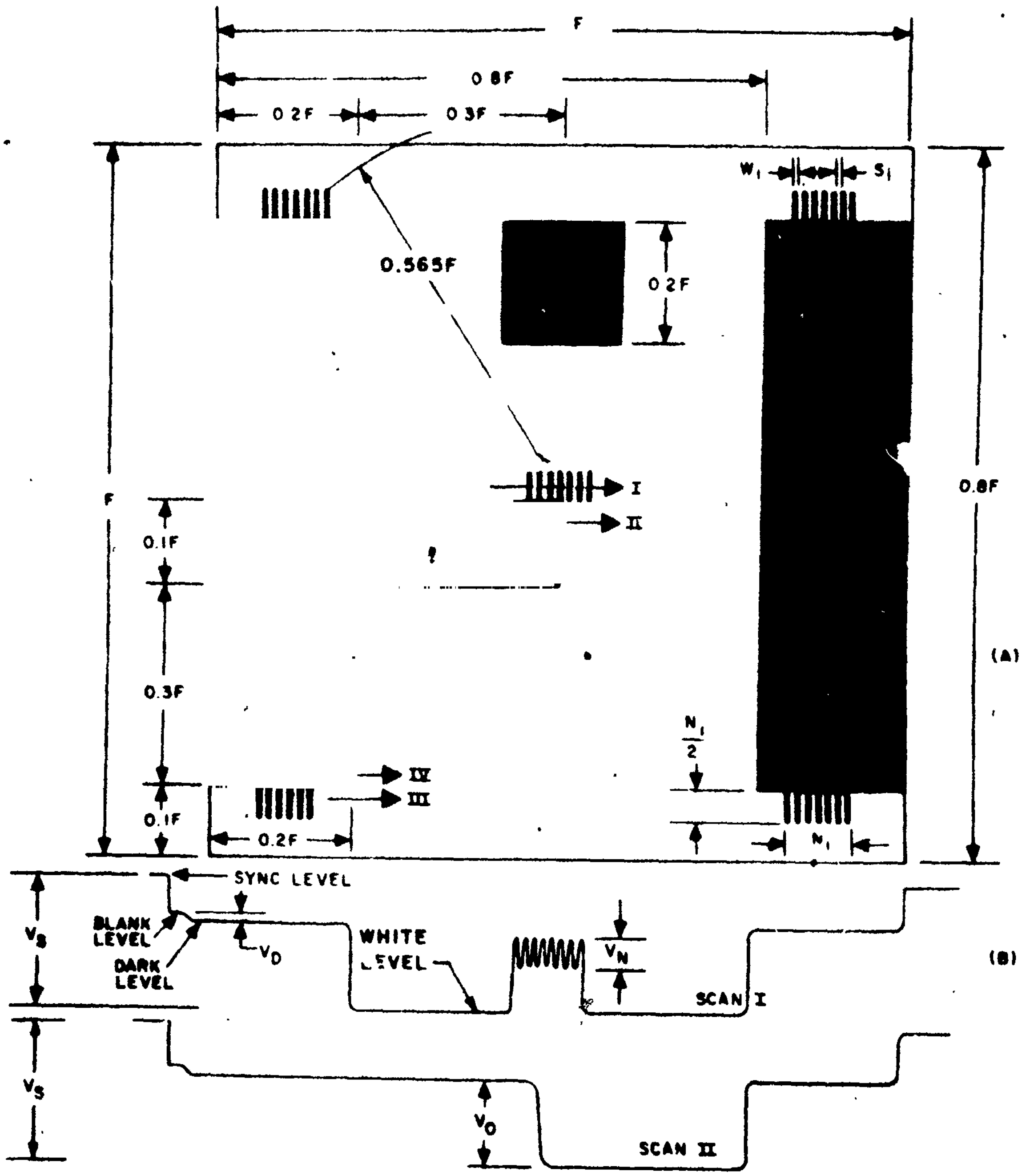
For test purposes the relative response shall be taken as

$$r_H(N) \equiv \frac{V_N}{V_c}$$

where V_N and V_c are defined as shown in Figure 5-1(B) which is the video voltage during a single horizontal sweep across a nominal, high contrast test target sketched in Figure 5-1(A). V_N is the average peak to peak video voltage at the line frequency N. N is the total number of black and white lines required to fill the frame. Equivalent test targets may be used so long as the line frequency used to determine V_c is N = 5 or less and at least 10 black and 10 white lines, at the line frequency N, are averaged.

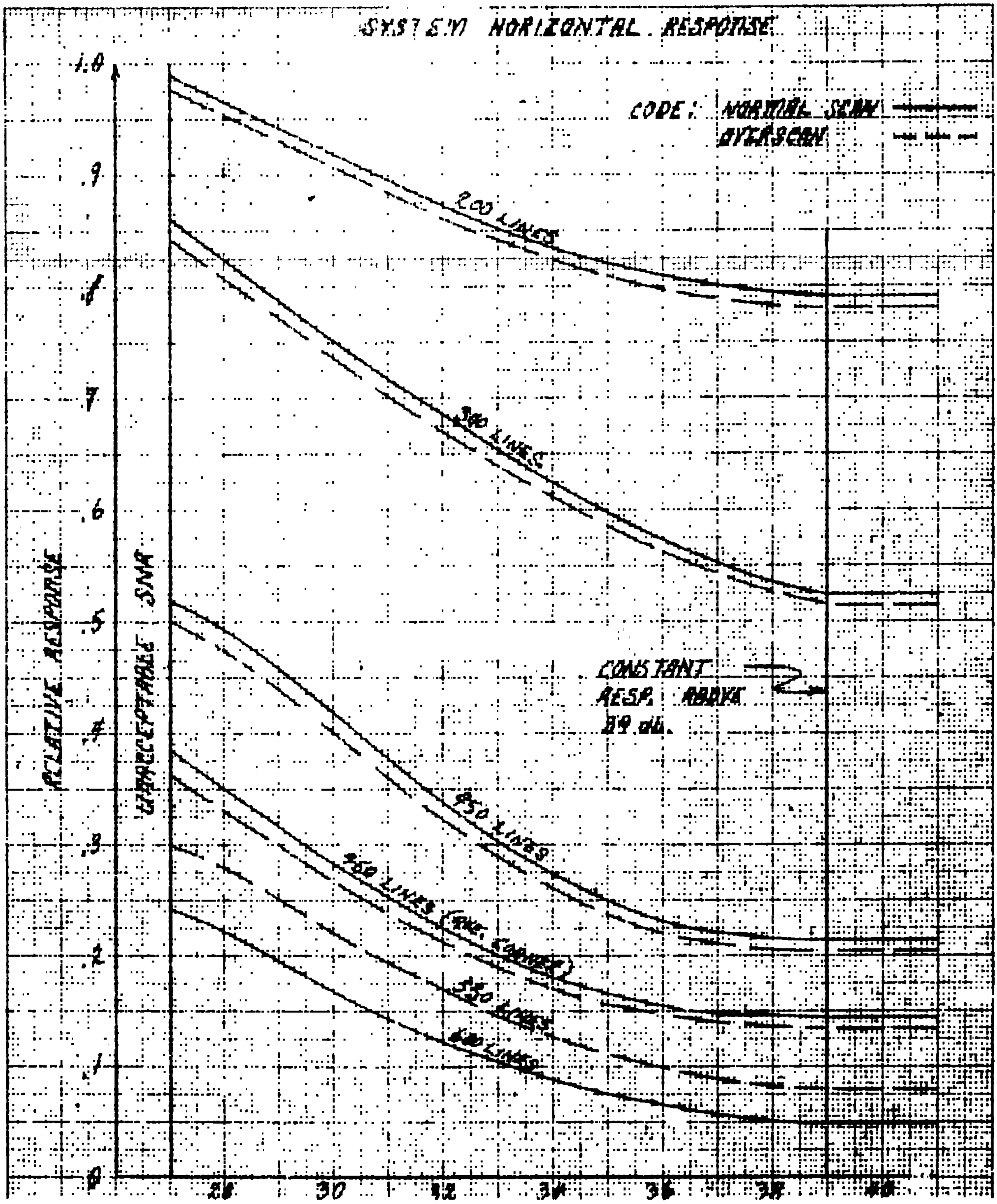
Resolution tests shall be performed at highlight exposures not exceeding the "effective saturation exposure." This shall be defined as that exposure level above which the system gamma becomes less than 0.7, where gamma is the slope of log output signal plotted against log exposure.

Acceptable evidence that the response of Figure 5-2 has been achieved is that the measured response equals or exceeds the response specified at 450 lines and at any two of the other three line frequencies at the corresponding measured signal-to-noise ratio.



F = FRAME SIDE
$S_i = W_i$ = LINE WIDTH = $\frac{F}{R_i}$ WHERE
R_i = 200, 300, 450, OR 600 LINES
N_i = 19 F/R_i
SCANS III AND IV FOR CORNER RESPONSE

Horizontal Resolution Test Chart
Figure 5-1



Signal to Noise Ratio P-P/RMS (db)

Figure 5-2

5.2.2.6 Emergency Mode System Horizontal Relative Response (Resolution)

As defined in 5.2.2.5, the center of frame emergency mode relative response, for spacecraft 1 thru 3 shall be 0.4 or higher at N = 200 lines (0.55 milliradian per TV line in a narrow angle and 2.2 milliradian per TV line in wide angle) concurrent with a signal-to-noise ratio as specified in 5.2.2.10 the center of frame emergency mode relative response for spacecraft 4 thru 7, shall be 0.37 or higher at N = 200 lines commensurate with the above specified SNR.

5.2.2.7 Normal Mode System Vertical Relative Response (Resolution)

The square wave relative response, $r_V(N)$ shall equal or exeed that shown in Figure 5-2. For test purposes the relative response will be taken as

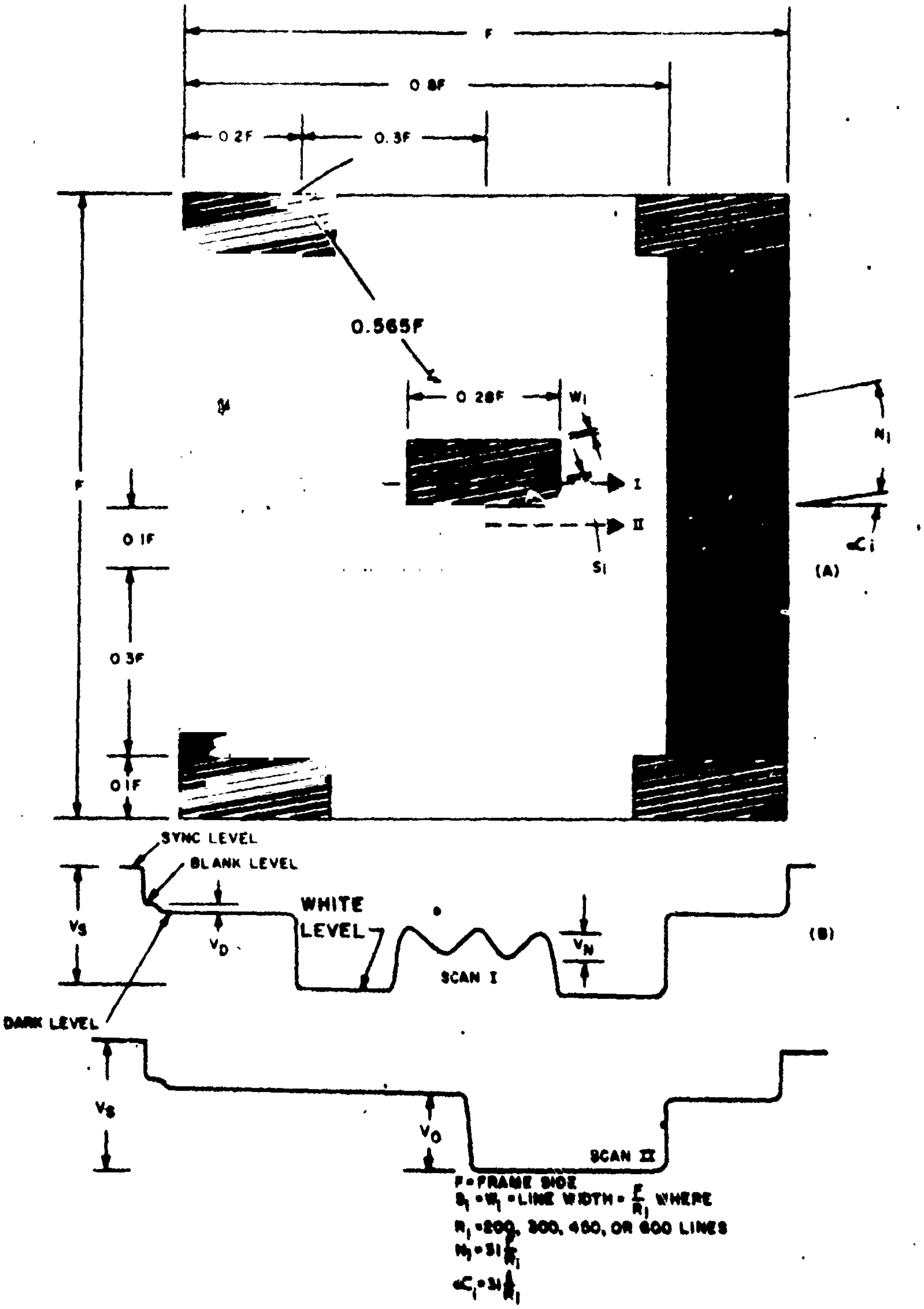
$$r_V(N) = \frac{V_N}{V_O}$$

where V_N and V_O are defined as shown in Figure 5-3(B), which is the video voltage during a single horizontal sweep across a nominal high contrast test target as sketched in Figure 5-3(A). V_N is the average peak-to-peak video voltage at the line frequency N. Test targets shall be designed so that the line frequency used to determine V is N = 5 or less and at least 4 black and 4 white lines, at the line frequency N, are averaged. The tilt angle shall be minimum and shall be such that the resultant video frequency does not exceed 12 kc. The video shall be filtered to a 3 db bandwidth of 20 kc with less than 0.5 db loss at 12 kc for test purposes.

Acceptable evidence that the response of Figure 5-2 has been achieved is that the measured response equals or exceeds the response specified at 450 lines and at any tow of the other three line frequencies shown in Table 2. Vertical relative response test values for corner relative response shall be averaged as in 5.2.2.5. To insure that the vertical resolution is not unduly degraded by the raster, at least 550 active lines shall be present throughout the range of operating environment.

TABLE 2

	SC1-4 N	SC1-4 r_V	SC5-7 N	SC5-7 r_V
Center	200	.6	200	.6
	300	.35	300	.35
	450	.15	450	.15
	600	.06	550	.09
Average Corner	450	.08	450	.08
Minimum Corner	450	.04	450	.04



TITLE	NUMBER
FIGURE 5-3 VERTICAL RESOLUTION TEST PATTERN	244510
SURVEYOR LABORATORY HUGHES AIRCRAFT COMPANY SPACE SYSTEMS DIVISION	REVISION H / SC PAGE 127 OF 177 PAGES

5.2.2.8 Emergency Mode System Vertical Relative Response (Resolution)

The center of frame emergency mode vertical resolution shall be 0.4 or higher at N = 200 lines (0.55 milliradians per TV line in narrow angle and 2.2 miliradians per TV line in wide angle) as defined in 5.2.2.7 with the exception that the tilt angle shall be such that the resultant video frequency does not exceed 200 cps.

5.2.2.9 Normal Mode System Signal-to-Noise Ratio

The peak to peak signal to rms noise ratio, as defined below, shall be 27 db or greater, including vidicon, video amplifier and receiver noise under laboratory ambient conditions, and consistent with the relative response requirement of 5.2.2.5. Up to 4 db degradation at the extremes of the operating temperature range is acceptable. Under conditions of normal vidicon exposure and monitor adjustment, no periodic electronic noise shall have peak-to-peak amplitudes greater than 8% of V_s (see Figure 5-1), nor shall other interference be discernible on the reproduced picture. For test purposes the signal-to-noise ratio shall be defined as

$$S/N = 20 \log \frac{V_o}{\sigma}$$

where V_o is defined as in Figure 5-1(B). The rms noise value shall be determined by the following method or one of equal precision. Under normal scanning operation photographic or other records shall be made of the video voltage with the lens covered so that no light is admitted. The maximum and minimum voltage levels which are exceeded by 10 per cent of the total maxima and minima, respectively, which are contained in the noise record, shall be measured and the voltage difference taken as 4σ. The time scale of the noise waveform as displayed on the CRO shall be such that approximately 15-20 extrema are displayed per centimeter. At least 1 millisecond of record is required to provide an adequate statistical sample.

5.2.2.10 Emergency Mode System Signal-to-Noise Ratio

The resolution requirement of 5.2.2.6 shall be met concurrently with a signal-to-noise ratio of 17 db, as defined in 5.2.2.9. At least 180 milliseconds of record are required to provide an adequate statistical sample.

5.2.2.11 Normal Mode Luminance Range

The requirements of 5.2.2.5 and 5.2.2.9 shall be met in the shuttered mode at f/4 for highlight luminance of between 50 and 150 foot lamberts, and at f/22 with 32 times that highlight luminance used at f/4. +The operating range shall be extended to at least 6000 ft lamberts by employing

+When color filters are supplied by JPL, HAC should attempt to insure and verify that this requirement is met by coordination with JPL.

one of the color filters. The luminance value shall be taken as that of a nominal 5300 K source, as measured by a brightness meter whose spectral response matches the CIE standard relative luminous efficiency curve.

5.2.2.12 Emergency Mode Luminance Range

The requirements of 5.2.2.6 and 5.2.2.10 shall be met for a highlight luminance not exceeding 300 foot lamberts with an f/4 iris setting while in the shuttered exposure mode.

5.2.2.13 Normal Mode Dynamic Range

Throughout the luminance range of 5.2.2.11 and concurrent with the requirements of 5.2.2.5 and 5.2.2.7 the system shall provide a dynamic range of 32:1 which may be evidenced by the ability to distinguish at least 11 gray levels spaced at a ratio of $\sqrt{2}$ in luminance, assuming any previous image is completely erased.

5.2.2.14 Emergency Mode Dynamic Range

Throughout the luminance range of 5.2.2.12 and concurrent with the requirements of 5.2.2.6 and 5.2.2.8 the system shall provide a dynamic range of 11:1 which may be evidenced by the ability to distinguish at least 8 gray levels spaced at a ratio of $\sqrt{2}$ in luminance, assuming any previous image is completely erased.

5.2.2.15 Veiling Glare-Stray Light

Insofar as possible the effects of bright objects and bright extended areas in the field of view of the mirror and lens (within and outside the frame) shall be minimized. In order to assist sequence planning, system tests shall be performed to determine possible restrictions on camera operation.

5.2.2.16 Shading

Including all factors (vidicon, shutter, lens, mirror, transmitter drift, etc.) peak-to-peak variation in video output as a function of position in the format, when the camera is exposed to a uniformly illuminated field, shall be less than 50 (4) percent of center of frame video voltage over at least 75 percent of the format, and less than 65 (4) percent of center of frame video voltage over the rest of the format.

5.2.2.17 System Linearity

The system linearity shall be such that after correction for lens distortion and for linear reseau distortion, the corrected positions of image points shall correspond to their undistorted image positions within 1 per cent (3) of the frame side for an object consisting of a plane grid perpendicular to the line of sight. The undistorted image referred to above shall be scaled-down replica of the object, for which the scale factor shall be determined during post-data analysis by a least squares fit of at least twenty widely spaced object-replica points with the corresponding points in the image. System linearity shall be measured in the normal (600 line) mode, using the 100 mm focal length, and demonstration of the required linearity at one focus step position shall be considered acceptable evidence that the above accuracy is achieved.

5.2.2.19 Operating Environment

The system shall be capable of meeting all performance requirements for sun angles less than 85 degrees from the zenith (approximately 25 hours from the terminator at 65 degrees latitude) assuming sun illumination of the mirror hood and the immediate surface area around the spacecraft. The system shall be capable of meeting all performance requirements for zenith-sun angles less than 75 degrees when the camera is in shadow. This capability shall be provided for all selenographic latitudes between 65 N and 65 S.

5.2.2.20 Minimum Operating Time

The system shall be capable of meeting all performance requirements for at least 0.8 hour of operation during any 8 hour period within the conditions of 5.2.2.19. Sun angles less than 15° from zenith may be excluded.

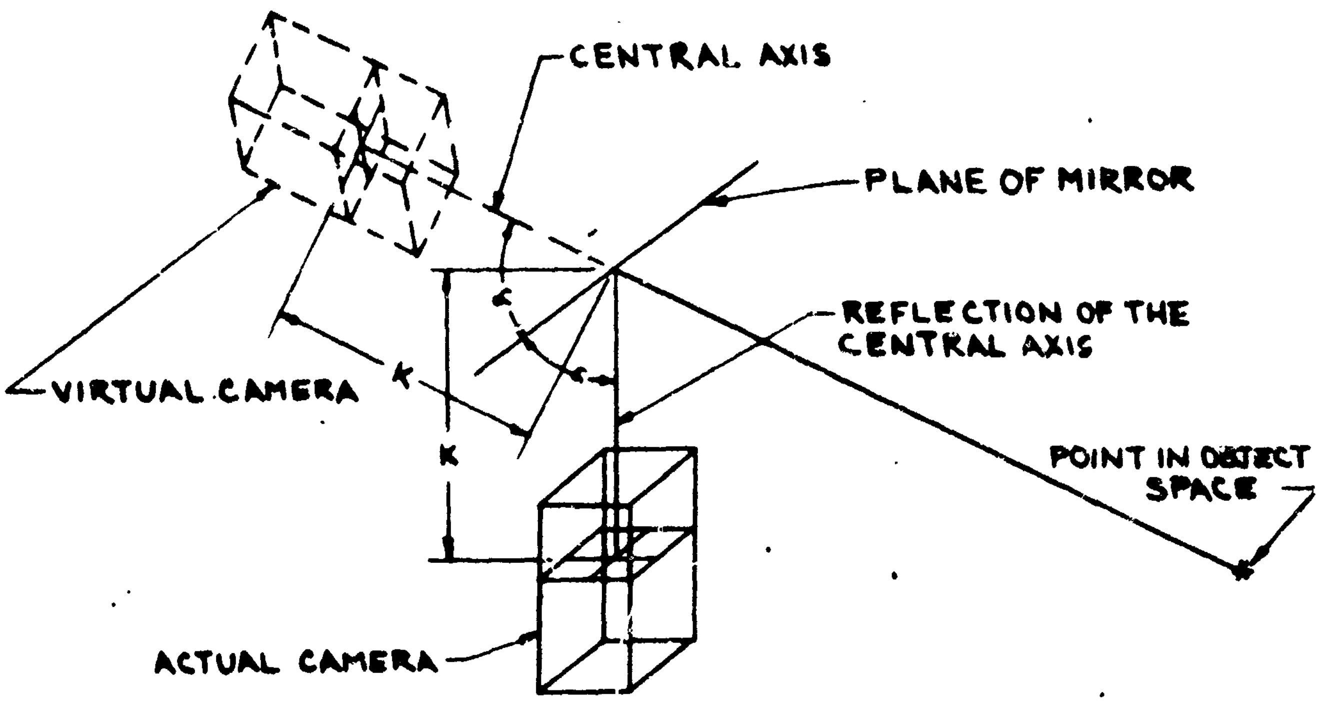
5.2.2.21 Warm Up Time

When completely shadowed, the camera shall be capable of meeting all performance requirements under the following conditions:

a) Within 5 hours (nominally) or within 10 hours (minimum voltage conditions) after temperature control is initiated any time while the spacecraft is in transit.

b) Within one-half hour after touchdown at any of the selenographic locations described in 5.2.2.19 provided that temperature control is initiated during transit 5

**The virtual camera is defined here as the image, or projection, behind the mirror, of the real lens plus vidicon system (see Figure 5-4). See 5.2.6.4 for the definition of central axis as used in Figure 5-4.



TITLE	NUMBER
FIGURE 5-4 CENTRAL AXIS LOCATION	224510
SURVEYOR LABORATORY HUGHES AIRCRAFT COMPANY SPACE SYSTEMS DIVISION	REVISION N · MODEL SC

hours (nominally) or 10 hours (minimum voltage conditions) prior to touchdown and maintained, if required by the landing location, after touchdown.

c) Within 5 hours (nominally) or within 10 hours (minimum voltage conditions) after temperature control is initiated at any time after touchdown at any of the selenographic locations described in 5.2.2.19.

5.2.2.23 Night Survival

Assuring heating is initiated at least 10 hours before touchdown (minimum battery voltage) and continued after touchdown, all mechanical and electrical functions of the camera shall operate for at least 10 hours after night touchdown.

5.2.2.24 Line-to-Line Jitter

The Line-to-line jitter performance is primarily a function of the ground equipment. CDC performance is specified in 6.2.4.1.

5.2.2.25 Frame Identification

5.2.2.25.1 Frequency of Transmittal: Identification of all commandable camera settings shall be transmitted with each frame.

5.2.2.25.2 Discrete functions:. The following discrete functions shall be a part of the frame identification information:

a) Camera Number - This attribute indicates which, if any, camera has been energized.

b) Shutter Mode - This attribute indicates whether the camera is set to operate in open shutter mode or in normal (150 MS) shutter mode.

c) Focal Length Setting - This attribute indicates whether the camera lens is set to operate in wide angle or narrow angle mode.

d) Lens Iris Servo On/Off - This attribute indicates whether the camera iris servo is energized. When energized, iris servo will position the lens iris automatically; when not energized, lens iris may be adjusted by means of ground command.

e) Multiple Step Focus On/Off - This attribute indicates whether the multiple step focus circuitry is energized or not.

f) Filter Wheel Position - This attribute indicates the quadrant of the filter wheel which is positioned in the optical path.

g) The telemetered accuracy of the discrete position, of each of the above functions, shall be such that the position shall be known(4).

5.2.2.25.3 Temperature Functions

The following temperatures shall be a part of the frame identification information:

a) Camera electronics temperature

b) Vidicon faceplate temperature

The actual temperature of each of these locations shall be determinable to an accuracy of ± 10°F at the input to the A/D converter.

5.2.2.25.4 Position Information

The following position information shall be a part of the frame identification information and shall be known to the following accuracy:

a) Mirror Azimuth - The achieved azimuth step shall be known with an uncertainty not exceeding 1 part in 100 using both position readout data and transmitted command information. Additionally, position of each azimuth step shall be calibrated and known according to the requirements of paragraph 5.2.3.24.

b) Mirror Elevation - The achieved elevation step shall be known with an uncertainty not exceeding 1 part in 100 using both position readout data and transmitted command information. Additionally, position of each elevation step shall be calibrated and known according to the requirements of paragraph 5.2.3.24.

c) Focus Setting - The uncertainty in knowledge regarding the actual focus position shall not exceed 1/4 focus step.

d) Iris Setting - The uncertainty in knowledge regarding the actual iris setting shall not exceed five percent of the iris area at the f/4 mechanical end stop and at nominal settings of f/5.6; f/8; f/11; and f/16, inclusive.

5.2.2.25.5 Bit Error Probability

The frame ID PCM data shall be recovered with a bit error rate $\leq 3 \times 10^{-3}$, at a carrier predetection SNR of 11.5 db, rms-to-rms, in the normal mode.

5.2.3 Camera Subsystem

The camera subsystem is an integrated assembly which contains as major functional elements the vidicon, lens, filter wheel, video amplifier, sweep and sync circuits and mirror assembly. The performance requirements of this section are established to insure that the combination of the camera and other system elements as previously defined meets the system requirements of 5.2.2. In cases where no corresponding requirement appears in this section, it is assumed the performance degradation in other elements is negligible and that the camera performance equals or is consistent with the system requirement.

5.2.3.1 Elevation Coverage

Elevation coverage shall be from -58 degrees with up to 50 percent vignetting (mirror only) to +26.5 degrees with no vignetting (mirror only) with respect to the mirror azimuth plane.

5.2.3.2 Normal Mode Camera Horizontal Relative Response

After appropriate filtering the camera video output shall meet the response and commensurate signal-to-noise ratio as shown in Figure 5-5. No corner relative response shall be less than 50 percent of the required average corner relative response.

5.2.3.3 Emergency Mode Camera Horizontal Relative Response

Camera video output shall meet the requirements of 5.2.2.6.

5.2.3.4 Normal Mode Camera Vertical Relative Response

Camera video output shall meet the requirements of 5.2.2.7.

5.2.3.5 Emergency Mode Camera Vertical Response

Camera video output shall meet the requirements of 5.2.2.8.

5.2.3.6 Normal Mode Camera Signal-to-Noise Ratio

The peak-to-peak signal to rms noise ratio shall be 28 db or greater, as defined in 5.2.2.9 and shall be consistent with the horizontal relative response as shown in Figure 5-5. No periodic electronic noise shall have peak-to-peak amplitudes greater than 8 percent V_s (see Figure 5-1).

5.2.3.7 Emergency Mode Camera Signal-to-Noise Ratio

The resolution requirement of 5.2.3.3 shall be met concurrently with a signal-to-noise ratio of 30 db, or greater. No periodic electronic noise shall have peak-to-peak amplitudes greater than 8 percent of V_s (see Figure 5-1).

5.2.3.8 Normal Mode Sensitivity

The requirements of 5.2.3.2 and 5.2.3.6 shall be met for the luminance conditions of 5.2.2.11.

5.2.3.9 Emergency Mode Sensitivity

The requirements of 5.2.3.3 and 5.2.3.7 whall be met for luminance conditions of 5.2.2.12.

5.2.3.10 Camera Scanning Linearity

The maximum deflection of the scanned raster edge, in normal mode, from a straight line, due to the effects of magnetic deflection and sweep waveform nonlinearities, shall be no greater than 16 TV lines. Variations in scanning speed over the entire format shall be no greater than 2 percent of the nominal scanning speed. All other effects such as line-to-line jitter, power supply loading, circuit switching, circuit intercoupling, etc., shall not cause distortion greater than ±4(4) TV lines on a straight line less than 150 lines in length. Additional fixed vidicon distortion shall not exceed 6 TV lines between any two adjacent exterior reseau marks and shall be known to 2 lines.

5.2.3.11 Filter Wheel

The camera shall contain a 4 position filter wheel capable of accepting color or polarizing elements. In the case of the latter, the direction of the plane of polarization shall be known to ±3 degrees (2). The time to step from one filter position to the next shall not exceed 1 second. Each filter wheel step position shall be 90 degrees ±6 degrees(4). Suitable end stops shall be provided such that no more than 4 steps can be taken in either direction. The clear window and color filters shall be arranged in a counterclockwise direction, as seen from the top (hood) view in the filter wheel in the following order: clear window, green filter, blue filter, red filter. The clear filter shall be placed in the filter wheel sector that is over the lens when the filter wheel has been set to its maximum counterclockwise direction, as seen from the hood.

5.2.3.12 Raster

The scanning raster shall contain 600 ±50(4) active lines in normal mode and 200 ± 10(4) active lines in emergency mode.

5.2.3.13 Signal Output

The peak-to-peak video voltage at low frequency input shall not be less than 65 percent of the full signal range ($V_o/V_s \geq 0.65$ in Figure 5-1(B)) for an effective saturation exposure (as defined in 5.2.2.5 and with a 5300 K source) in normal mode and 20 percent of the full scale range in emergency mode. With the aperture adjusted to produce 65 percent video voltage, the sum of dark voltage (V_d) plus video voltage (V_o) shall not exceed 85 percent of the full signal range, nor cause amplifier saturation throughout the vidicon temperature range from -20°F to +130°F.

5.2.3.14 Vidicon Spectral Response

Spectral response shall be such that an effective saturation exposure as defined in 5.2.2.5 from a 5300 K source will produce at least 9×10^{-9} amperes desired, 3.2×10^{-9} amperes required, using the filters, Corning CS-4-65 (green), Corning CS-5-57 (blue), and 2.4×10^{-9} using the Corning CS-3-66 (red) filter.

5.2.3.15 Persistence

The erasure characteristics shall be such that when "black" areas are superimposed on "white" areas in successive frames of a normal survey sequence, the black area video voltage differs from the black area video voltage for a sustained sequence on a stationary pattern by no more than 6 percent of the maximum black to white voltage difference. For purposes of this requirement, a white to black exposure ratio of 30 with the white exposure equal to the effective saturation exposure (see 5.2.2.5) is assumed. It is desired that this persistence be no more than 1 percent.

5.2.3.16 Overexposure

A single exposure anywhere in the frame of up to 16 times the effective saturation exposure shall not cause picture degradation, after seven erase scans, in subsequent pictures in a sequence except as specified in 5.2.3.15.

5.2.3.17 Shading

"Peak-to-peak variation in vidicon output voltage as a function of position in the format, when exposed to a uniform exposure, with a SNR as specified in 5.2.3.6, on the vidicon face shall not be greater than: 1) 25 percent (30 percent at lens f/no. =4) of the center of frame output voltage for the average found in I; 2) 35 percent (50 percent at lens f/no. =4) of the center of frame output for the average found in II.

I. Average the four following output voltages:

a) Center of trace output voltage for the 6% from the top of format scan line

b) Center of trace output voltage for the 6% from the bottom of format scan line

c) Two output voltages at the 6% from edge of trace for the center of format scan line

II. Average the two following output voltages:

a) Two output voltages at the 6% from edge of trace for the 6% from the top format scan line

b) Two output voltage at the 6% from edge of trace for the 6% from the bottom of format scan line"

5.2.3.18 Reseau

A reseau, capable of correcting the geometric distortion in the TV system after the lens to an accuracy sufficient to meet the requirements of 5.2.2.17, shall be inscribed on the inside of the vidicon face. Maps shall be made of the inscribed reseau such that the position of reseau points shall be known in two directions to within ± 5 microns[4], in an appropriate (x, y) coordinate system.

5.2.3.19 Dark Current Calibration

Dark current calibration not required for A21.

5.2.3.20 Mirror Step Sizes

Including all dispersions from step to step and the reproducibility of each step, mirrors shall have step sizes of 2.48 ± 0.1 degree (4) in elevation, 3.0 ± 0.1 degree (4) in azimuth.

5.2.3.21. Mirror Step Datum

The datum for orientation of the elevation steps shall be such that the center of a narrow angle frame may be aimed at -42.9 ± 2.0 degree (4), with respect to camera coordinates. The datum for orientation of the azimuth step shall be such that the center of a frame may be aimed perpendicular to the camera face, within 2.0 degrees (4). The camera face is the flat section of the camera housing containing the electrical connectors.

5.2.3.22 Mirror Azimuth Stop

A suitable stop shall be provided in azimuth to inhibit the mirror from stepping beyond 132 degrees in a counterclockwise direction and 222 degrees in a clockwise direction, measured from the azimuth datum line (see 5.2.3.21), on the azimuth stepping circle, as viewed from above the camera.

5.2.3.23 Mirror Rotation Axis Alignment

The axis of rotation of the mirror shall intersect the image plane - for infinity focus - within ___ inch of the format center. This axis shall be parallel to the reflection of the central axis of the camera within ±0.1 degree(4) (see figure 5-4, and see 5.2.6.4 for the definition of the central axis of the camera). The elevation axis of the mirror shall pass through the mirror surface within .005 inch(4), and through the azimuth axis within .010 inch(4). The elevation and azimuth axes shall be perpendicular within 2 minutes of arc.

5.2.3.24 Mirror Stepping Accuracy

Actual mirror step positions shall be calibrated and shall be known and reproducible to 6.0(4) minutes of arc with respect to the mirror mounting regardless of the direction of rotation.

5.2.3.25 Mirror Stepping Rate

The mirror stepping rate capability listed below shall apply for electronics temperatures between -20°F. and +165°F., mirror assembly temperatures between -50° and +165°F., unregulated 22 volt (nominal) supply voltage between 16.9 and 27.5 volts and stepping pulse width less than or equal to 150 ms.

a) Continuous elevation stepping at a rate of 2 steps per second for sustained sequences of steps.

b) Continuous azimuth stepping at a rate of 4 steps per second for sustained sequences of steps.

5.2.3.26 Mirror Settling Time

After stepping through 2 steps, mirror vibrations shall be damped to ±0.1 minute of arc(3) amplitude within 0.5 second.

5.2.3.27 Mirror Optical Quality

The mirrors as mounted in the camera environment on the lunar surface, shall conform to the following:

a) Powers: maximum of 2 fringes per inch.

b) Irregularity: maximum of 1 fringe per inch.

c) Surface Quality: equivalent to 60-30 per MIL-O-13830.

d) Average Specular Reflectivity: At least 86 percent (within ±60 degrees from the normal) between 4000 and 7000 angstroms.

e) Average Diffuse Reflectivity: 0.25 percent maximum required, 0.03 percent maximum desired, between 4000 and 7000 angstroms.

5.2.3.28 Cleanliness

The mirror shall be capable of being stowed for dust protection. Prior to launch the lens, filter wheel, and the mirror shall be cleaned, all dust particles shall be removed from the mirror surfaces, and then the mirrors shall be stepped to the closed hood position.

5.2.3.29 Lens Resolution

The minimum average resolution shall be equal to or greater than 70 line pairs per mm over the entire image and for all object distances for 25 mm and 100 mm focal-lengths, over the range 0.4 to 0.7 micron spectral region, with object distances from 6 feet to infinity. At a 4 foot object distance the minimum average shall be equal to or greater than 60 line pairs per mm. No individual value shall fall below 20 line pairs per mm at either focal length or at any object distance. The above values shall be met with each of the 3 primary color filters+ of the filter wheel, placed before the lens, one at a time, without refocusing.

5.2.3.30 Lens Distortion

The overall distortion due to lens, filter, and reflex viewer shall be no greater than 4 percent(4) for 25 mm and 100 mm focal-lengths over the vidicon format. This distortion shall be calibrated to ±0.5(4) percent of the lens semi-diagonal.

5.2.3.31 Focal Length

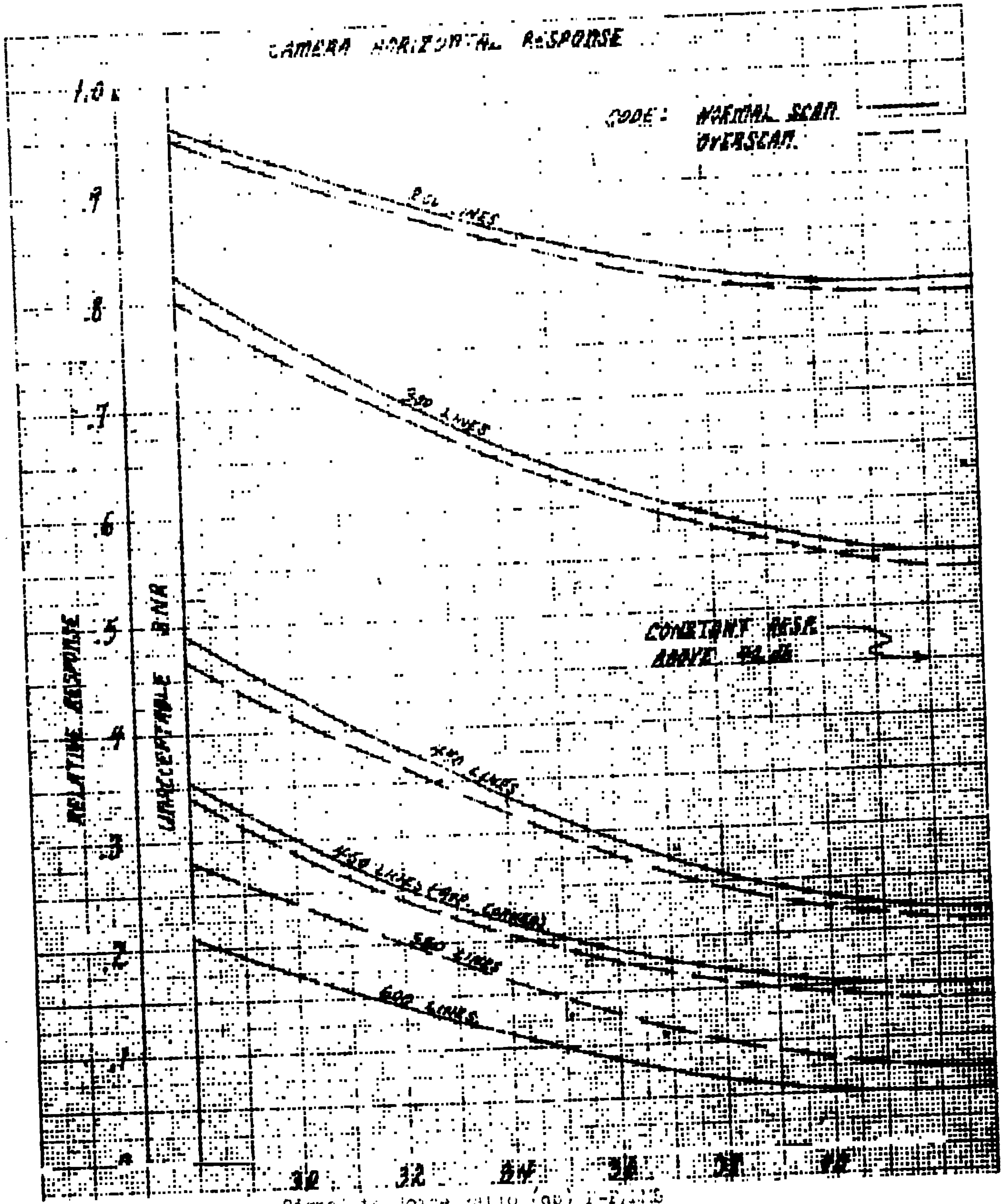
Focal length shall be commandable to 100 ±2 mm and 25 ±0.5 mm at two separate end stop positions. The focal length at each end stop position shall be calibrated to an accuracy of 0.25 percent at factory ambient conditions. Repeatability of focal length end stop positions for thermal vacuum environments shall be determined.

5.2.3.32 Iris

The iris shall have the capability of operating in the following modes.

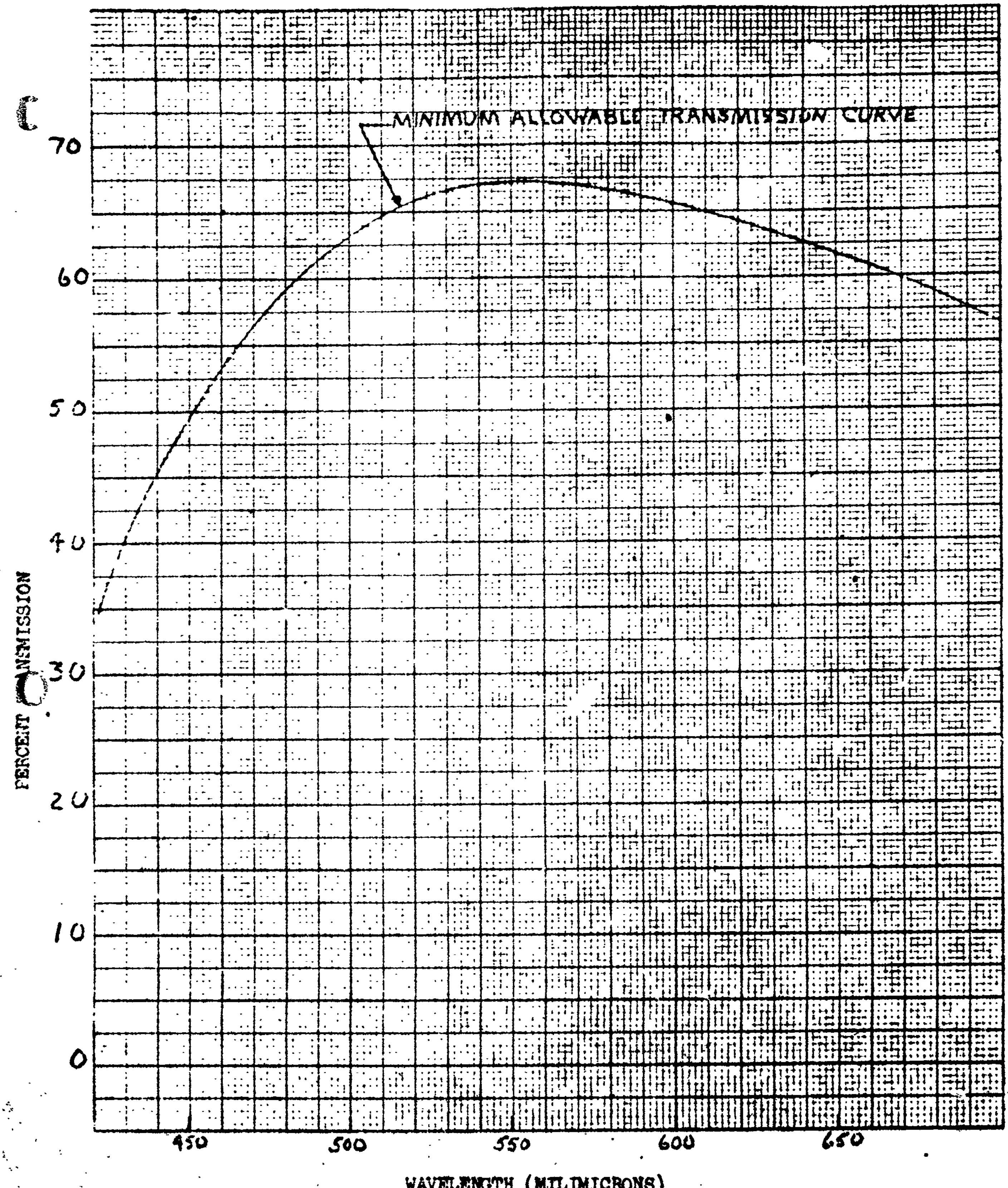
Command Mode: The lens aperture shall be command adjustable to f/4, f/5.6, f/8, f/11, f/16 and f/22, to within ±20(4) percent of the total area of a specified operation for any aperture from f/4 to f/22.

+When color filters are supplied by JPL, HAC should attempt to insure and verify that this requirement is met by coordination with JPL.



Signal to Noise Ratio (dB) P-P/RMS

Figure 5-5



TITLE	NUMBER	
FIGURE 5-6 LENS TRANSMISSION	224510	
SURVEYOR LABORATORY HUGHES AIRCRAFT COMPANY SPACE SYSTEMS DIVISION	REVISION H	MODEL SC
	PAGE 142 OF 177 PAGES	

Automatic Mode: On command, a light sample shall be employed to control the iris automatically to the average of the observed scene brightness. In this mode, the iris shall correct and settle within 0.5 second after a change in scene brightness corresponding to one f/stop. The iris shall correct to within 40 percent[4] of the desired average vidicon illuminance. The automatic iris shall be calibrated and adjusted using a test chart having an average contrast ratio of 2.5 to 1.0 and illuminated by a 5300 K source. The automatic iris under the above conditions shall be adjusted such that the observed test chart highlights produce an effective saturation exposure (see 5.2.2.5).

5.2.3.33 Transmission

On axis light transmission at each wavelength through the lens assembly including beam-splitter and lens shall be no less than as shown in Figure 5-6.

5.2.3.34 Lens Vignetting

Image illumination across the format, for uniform input, shall be at least 60 percent of that in the center of the format at any iris setting.

5.2.3.35 Focus Distance

The focus distance shall be adjustable in not less than 50 steps from four feet to infinity at either focal-length. The focus distance at any step is to be the same at 100 mm focal-length as at 25 mm focal-length. Focusing accuracy shall be such that at any step the absolute difference between the actual focus distance and the nominal distance shall not exceed 50 percent[4] of the difference between the nominal focus distance and the next lower nominal focus distance.

5.2.3.36 Focus Stepping

The following two command modes shall be provided to change object distance:

a) Single focus step commands which increase or decrease the focus position by one of the 50 or more focus steps. The survey TV subsystem shall be capable of accepting and reacting to single focus step commands transmitted at the rate of one command every 0.5 seconds or slower. Single step and mechanical settling time shall not exceed 0.3 second.

b) Multiple step focus commands which increase or decrease the focus position until either the command is repeated or an end stop is reached. After reaching an end stop, the multiple step focus command shall be repeated to stop the drive motors. When activated by the multiple step focus command, focus position shall vary at the rate of 4.0 ± 1.0 steps per second. The survey TV subsystem shall be capable of accepting and reacting to multiple step focus commands transmitted at the rate of one command every 0.5 seconds or slower. Mechanical settling time after the last step has been achieved shall not exceed 0.1 second.

5.2.3.37 Shutter

The shutter shall have two modes of operation:

a) Shutter time of 150 ± 10 msec(4)

b) Open shutter mode.

5.2.3.38 Camera Life

The camera shall be designed to provide a minimum of fifty-thousand(4) TV frames.

5.2.3.39 Camera Sync Pulse Rise Time

The rise time (10% - 90%) of horizontal sync pulse shall be equal to or less than 0.5 microseconds at the output of the camera sync circuitry in normal mode and 75 microseconds in emergency mode.

5.2.3.40 Camera Sync Pulse Frequency Stability

Variations in horizontal sweep pulse-to-pulse period shall not exceed ± .5 μsec(4), in normal mode, and ±400 μsecs in emergency mode. The sweep period shall be measured on the composite video waveform at the black (or dark current) level, for full scale light input to the vidocon, i.e., approximately 4 volts peak-to-peak video camera output.

5.2.3.41 Camera Mirror Hood

To meet the requirements of 5.2.2.15 the camera mirror hood shall be designed to prevent stray or reflected light from entering the lens. The interior of the hood shall be suitably finished to reduce to a minimum, reflections from its walls.

5.2.4 Signal Processing and Transmission

5.2.4.1 Modulator Response

The transmitter frequency deviation for sine wave inputs to the wideband FM video input of the transmitter shall be flat within ± 1.0 db from 1 cps to 220 kcps. The zero frequency transfer constant shall remain within ± 10% of its prelaunch value.

5.2.4.2 Transmission Bandwidth

Modulation gain shall be such that the composite video plus noise from the camera will not cause significant distortion or signal-to-noise ratio degradation due to deviations outside the bandwidths of 3.3 (-3 db) mc in normal mode and 10.5 kc (-2.5 db) in emergency mode.

5.2.4.3 Normal Mode System Demodulated Signal-to-Noise Ratio

Modulation parameters shall be such that a demodulated peak-to-peak signal to rms noise ratio of at least 43 db is obtained for a low frequency sinewave test signal having a peak-to-peak amplitude of 65% of the camera peak-to-peak composite video when demodulated in an ideal discriminator with a square passband 220 kc output filter and a predetection signal-to-noise ratios ≥11.5 db in a 3.3 mc square bandwidth defined in 5.2.4.4.).

5.2.4.4 Emergency Mode Demodulated Signal-to-Noise Ratio

Modulation parameters shall be such that a demodulated peak-to-peak signal to rms noise ratio of at least 20 db is obtained for a low frequency sinewave test signal having a peak-to-peak amplitude of 20% of the camera peak-to-peak composite video when demodulated in an ideal discriminator with a square passband 1.2 kc output filter and a predetection signal-to-noise ratio ≥5.5 db rms-to-rms in the 10.5 kc bandwidth defined in 5.2.4.2.

5.2.4.5 Spacecraft Transmission Rise Time and Overshoot

For a step output from the vidicon amplifier the rise time (10% - 90%) of the transmitter output frequency change shall be no greater than 1.0 microseconds in the normal mode, or 7 microseconds in the emergency mode. No more than 5% overshoot shall be present.

5.2.5 Post Landing Calibration Charts

Two charts shall be provided for post landing photometric, colorimetric, and resolution calibration.

5.2.5.1 Resolution

The chart shall include black and white bars of varying width sufficient in range to determine the post landing resolution, sensitivity and signal-to-noise performance of the camera. A contrast ratio of approximately 8 is required.

5.2.5.2 Mounting and Alignment

At least one chart shall be mounted in a plane normal within ± 5.0 degree to the line-of-sight of the camera when it views the chart for nominal landing conditions. The other chart shall be mounted such that it is viewable by the camera and in a plane as nearly as possible normal to the camera line of sight. The charts shall be contained within the inscribed circle of a narrow angle TV frame taken by the Surveyor TV camera when it views the chart for nominal landing conditions.

5.2.5.3 Environmental Requirement

The reflective properties of the charts shall be stable in the lunar environment and survive two lunar days and one lunar night contributing 2 percent or less error in photometry due to environmental effects exclusive of dust. The direction of the normal to each chart surface shall be known in S/C coordinate after landing to ± 1 degree (4) throughout the range of landing conditions specified in Section 3.11.

5.2.6 Camera Mounting and Alignment

5.2.6.1 Height

Mirror elevation axis shall be at least 4.5 feet above the nominal lunar surface. Maximum height consistent with 5.2.6.4 is desired.

5.2.6.2 Unobstructed Coverage

Maximum unobstructed azimuth coverage at all elevations is desired.

5.2.6.4 Alignment

Mounted orientation of the camera central axis shall be known for the mirror orientation before launch to ±0.5 degree(3) in spacecraft coordinates. The camera central axis is defined here as the line connecting the central reseau point of the virtual camera (see Figure 5-4) with an object which is imaged at the reseau point.

5.2.6.5 Camera Motion

Mounting shall be sufficiently rigid and/or damped that operation of any camera mechanical functions (e.g., mirror positioning) motion of the camera central angle (see 5.2.6.5) will be less than 10 sec of arc amplitude 0.5 second after the camera motion is complete. In addition, the mounting bracket shall be designed to insure the camera does not rotate or tilt within the bracket, due to launch and landing loads, to not more than ±0.1 degree(4) either in rotation, or in tilt, from its prelaunch settings.

5.2.6.6 Vidicon Format Alignment

One side of a TV camera vidicon format shall be aligned parallel to the spacecraft X axis within ± 1(4) degree.

5.2.7 Sequence Programming

5.2.7.1 Automatic Sequence

The high picture rate capabilities of the lunar TV system coupled with severe time and power restrictions require assurance that the ground environment will not limit the required TV surveillance function. There is, accordingly, a requirement for an automatic mode of operation, meeting the following requirements.

a) A prepared computer program for use by the SFOF computer in this program is designed to accept initial survey data and use the information to arrive at an optimum focusing program.

b) Photographic Analysis: The initial survey sequence requires that the rapid processed photographic frames be analyzed at an average rate of about 6 frames per minute. Thus, several people operating in parallel may be required to perform the data reduction and provide the relevant data as a computer input.

c) Tape Punching: Tape punching and format conversion equipment are required at the Goldstone DSS to convert the computer output to color coded aluminized mylar tape such that no delay of spacecraft sequence operation will result. The punched tape must be error checked before commands are transmitted from it.

All sequence repeats of the complete mapping sequence for the same focus settings will be run from the final corrected aluminized mylar tape if the iris servo is used; if specific iris stops are wanted for specific frames and/or if specific new focus settings (to achieve better resolution at points of particular interest) are desired for certain frames, the computer program should accept these as inputs and produce a new tape sequence program.

5.2.7.2 Manual Operation

The operator must be capable of manually commanding all camera modes from the console. This is necessary as a backup to the automatic mode, or in those cases where only a few isolated frames are wanted. It should be an objective to achieve as high a manual frame rate as possible.

5.3 Alpha Scattering Instrument:

5.3.1 General Requirements and Objectives

The Alpha Scattering Instrument will be used for compositional analysis of lunar surface materials. The instrument will detect backward scattered alpha particles as well as protons generated within the sample by the incident alpha particles.

5.3.2 Television Viewing

5.3.2.1 Stowed and Background Positions

It is desirable that television viewing of the sensor head be possible with:

a) The sensor head in the stowed position (on the standard sample)
b) The sensor head in the background count position.

5.3.2.2 Lunar Surface Position

All possible locations to which the sensor may be deployed shall be within the unobstructed view of the survey television camera. Television viewing capabilities shall include:

a) A narrow-angle survey of the lunar surface area to which the sensor head is to be deployed.
b) A wide-angle view of the sensor head on the lunar surface subsequent to deployment.
c) A narrow-angle survey of the sensor head on the lunar surface.

5.3.3 Environmental Operational Requirements

5.3.3.1 Operating Temperatures

5.3.3.1.1 Deployment Mechanism:

The Alpha Scattering Instrument Deployment Mechanism shall be capable of sensor head deployment when the temperature of the mechanism is within the range of -65°F and +275°F.

5.3.3.1.2 Alpha Scattering Instrument:

The Alpha Scattering Instrument shall be capable of operation when the following temperatures are concurrently realized:

a) The sensor head temperature lies within the range of from -40°F to +122°F.
b) The surface to which the Alpha Scattering Instrument Auxiliary (ASIA) and Electronics (ASIE) are mounted is within the range of from -4°F to +131°F.

5.3.3.2 Operating Environment

Thermal control shall be provided such that under conditions of continuous operation the temperature shall remain within the limits specified in 5.3.3.1 to within 3 hours of the day/night terminator for landings within ±65 degrees latitude (assuming use of the sensor head heater during the near terminator periods).

5.3.4 Instrument Survival Temperature

5.3.4.1 Sensor Head

The sensor head, including sensor electronics, is capable of withstanding without performance degradation non-operating temperatures between -300°F and +167°F.

Thermal control shall be provided such that the maximum non-operating temperature shall not be exceeded at any time during the lunar day.

5.3.4.2 Digital Electronics

The digital electronics shall be capable of withstanding without performance degradation non-operating temperatures between -130°F and +257°F.

5.3.4.3 Alpha Scattering Instrument Auxiliary (ASIA)

The ASIA shall be capable of surviving a non-operating temperature range of from -67°F to +185°F.

5.3.5 Heater Requirement

A heater shall be included in the sensor head so as to provide the capability of maintaining the sensor head temperature above -40°F during the near terminator periods of the lunar day. The heater shall be sized so as to be capable of maintaining the operating temperature level to within 3 hours from the day/night terminators.

5.3.6 Spacecraft Subsystem Data Characteristics

Signal	Type	Instrument Output Range	Accuracy Signal Processing Terminals to Transmitter Input
Alpha Particle Detector	digital	NA	Bit error rate 10^{-5}
Proton Detector	digital	NA	Bit error rate 10^{-5}
Sensor Heat Temp (from ESP)	analog	-300°F to +167°F	± 0.2% (4)
Electronics Temp (from ESP)	analog	-300°F to +257°F	± 0.2% (4)
Power Supply (+7 volt)	analog	0 - 5 V	± 1% (4)
Power Supply (+24volt)	analog	0 - 5 V	± 1% (4)
Guard Event Monitor	analog	0 - 5 V	± 1% (4)
Alpha Command Memory Verification	discrete analog	"on" 5 to 10v dc "off" 0 ± 1v dc	NA
Proton Command Memory Verification	discrete analog	"on" 5 to 10v dc "off" 0 ± 1v dc	NA

5.3.7 Signal Processing and Transmission

The alpha scattering experiment transmission link performance shall be such as to provide recovered alpha and proton count PCM data at a BER $\leq 1.1 \times 10^{-4}$, for minimum margin (worst case) multiplex predetection SNR conditions. The multiplex is defined as the simultaneous phase modulation of the RF carrier by the alpha count, proton count, and 550 BPS (0.3 radian modulation index) engineering data subcarriers. It shall be possible to transmit other engineering data rates, in this multiplex, with increased levels of alpha and proton count BER. It shall be presumed the spacecraft subsystems contribution to overall link BER is specified by para. 5.3.6.

The alpha particle and proton detector outputs shall each frequency modulate a subcarrier, as NRZ, PCM data, in a PCM/FM/PM mode, at 2200 BPS and 550 BPS, respectively. The remainder of the alpha scattering PCM engineering data, on an appropriate engineering data subcarrier, also in a PCM/FM/PM mode.

5.3.8 Service Life

The minimum service life objective of the instrument subsystem, excluding checkout and test time, shall be 45 hours (36 hours of operation plus 9 hours maximum interruption time). This corresponds to a spacecraft service life objective of 90 days.

6.0 GROUND EQUIPMENT PERFORMANCE REQUIREMENTS

6.1 DSIF Communication Link: A two-way communication link must be established between the earth and the spacecraft during transit and lunar operations. It is required that the spacecraft be compatible with the JPL Deep Space Instrumentation Facilities (DSIF) as defined in the DSIF Interface and Requirements Document. The document which defines the technical performance of the DSIF Mission Independent Equipment and its interfaces with the Surveyor CDC will be JPL EPD 260 "Surveyor Project/Deep Space Network Interface Agreement" Revision 1 (one) of 16 August 1965. The following DSIF capabilities and limitations are assumed in setting other dependent system functional requirements and design parameters:

6.1.1 Transmitter

6.1.1.1 Frequency: In 2110 mcs to 2120 mcs band.

6.1.1.2 Tunability: May be electronically tuned ±70 kc about center frequency. Center frequency may be placed in selected parts of the above band by changing crystals.

6.1.1.3 Frequency Stability: The long term (less than one day) stability shall be within 0.1 ppm. The short term stability (less than 1 min) shall be within 0.005 ppm.

6.1.1.4 Tuning Rate: The maximum rate of change of transmitter frequency during tracking shall be such that the rate of change of doppler frequency plus the transmitter frequency rate shall not exceed 1 kc/sec^2 for 30 sec.

6.1.1.5 Tuning Interval: The maximum time interval between transmitter tuning during transit shall be such that the transmitter is tuned before the doppler frequency has changed 75 kc from the frequency at the last tuning. The nominal tuning shall be such as to keep the received signal at the S/C within 1 kc (desired) of its open loop rest frequency.

6.1.1.6 Modulation: The transmitter shall be capable of being phase modulated for sending commands to the S/C.

6.1.1.7 R-F Power Output: It shall be possible to vary the transmitter power output from that level required for maximum distance operation to 20 db below that level.

6.1.2 Receiver

6.1.2.1 Frequency: In 2290 mc to 2300 mc band.

6.1.2.2 Tunability: May be electronically tuned ±70 kc about center frequency. Center frequency may be placed in selected parts of the above band by changing crystals.

6.1.2.3 Frequency Accuracy and Stability: The long term (less than one day) stability shall be within 0.1 ppm. The short term stability (less than one minute) shall be within 0.01 ppm.

6.1.2.4 Bandwidth: The receiver shall have an i-f bandwidth of at least 3.0 mc to receive television video signals.

6.1.3 Tracking Capability

6.1.3.1 Maximum Angular Tracking Rates: 0.7 deg/sec in hour angle and declination. (This limits DSIF capability for those parking orbit cases when injection takes place near the DSIF. In these "worst case" trajectories, angular rates up to 3.5 deg/sec will be experienced.)

6.1.3.2 Maximum Doppler Shift: Tuning range of the transmitter and receiver is limited to ±70 kc from their nominal; where the nominal frequency is determined by crystal selection. This tuning range, in conjunction with the spacecraft transmitter and receiver frequency uncertainties, results in a doppler coverage of up to approximately ±50 kc (one way). This limits the DSIF tracking capability for parking orbit cases where injection takes place near the DSIF. In these "worst case" trajectories, the doppler shift can be from +72 kc to -72 kc for the spacecraft receiver frequency (nominally 2113.3 mc) and from +78 kc to -77 kc for the spacecraft transmitter frequency (nominally 2295 mc). To accommodate these large frequency changes, it will be necessary to bias the transmitter and receiver center frequencies to a value such that tracking will be possible during only the last but major part of a pass.

6.1.3.3 Maximum Rate of Change of Doppler: Doppler rates up to 2 kc/sec^2 (two way) will be encountered under conditions wherein the angle rates do not exceed the values specified in paragraph 6.1.3.1. Greater doppler rates will exist but under those conditions, tracking will be limited by angle rate.

6.1.3.4 Loss of Lock: Momentary loss of lock by the DSIF receiver should be expected whenever the spacecraft transmitter power level is switched.

6.2 Command and Data Handling Console (CDC): This section provides the system functional requirements for the Surveyor ground command and data handling system as used at the various DSIF stations. The system referred to here is defined by the Specification 222c75. The CDC provides a command link to the spacecraft through the DSIF transmitter and receives information from the spacecraft through the DSIF receiver during a Surveyor mission. Provisions are also made to receive command data at the DSIF stations from the Space Flight Operations Facility (SFOF) at JPL for processing and subsequent transmission to the spacecraft and to also relay data from the spacecraft received at the DSIF stations to SFOF. Commands and data transmitted and received at the DSIF station may be stored in a recorder that is part of the DSIF station equipment or may be transferred to two computers which are part of the OSDP equipment.

6.2.1 Down-Link Telemetry Subsystem: The telemetry function consists of equipment which processes and restores the data contained on two signals furnished to the CDC from DSIF interfaces. With a received RF carrier from the S/C which is frequency modulated, a translated 10 MC IF signal from the DSIF receiver is sent to the CDC for demodulation. If the RF carrier is phase modulated, demodulation takes place in the DSIF receiver and the detected signal is sent to the CDC, for further subcarrier frequency demodulation. The telemetry data that can be accommodated in the CDC at any one time with the spacecraft transmitter in the PCM-FM-FM mode consists of 6 analog SCO (FM) channels, any one of 3 PCM SCO (FM) channels, and 1 SCO (FM) command enable/reject channel. The 6 FM channels supply information concerning gyro speeds and spacecraft acclerations when required. When the spacecraft TV is commanded off, the 6 FM channels may operate continuously. The selected PCM (commutated) FM channel of the three available channels is used in the CDC decommutator where the message frame impressed on the channel is divided into a maximum of 124 digital data words and 2 message sync words. The FM command enable/reject channel is connected to a display on the command console of the CDC to show the status of the CDC/spacecraft data link.

6.2.1.1 Wide Band Frequency Demodulator (Spec. No. 222245, Rev. E)

6.2.1.1.1 WB Television

a) Nominal Center Frequency: 10 MC

b) Maximum Peak Carrier Deviation: ±1.05 MC

c) Predetection Filter Bandwidth:
-3 DB± 0.5 DB Bandwidth is 3.3 MC; Equivalent noise bandwidth is 3.3 MC ±0.32 MC

d) Post Detection Video Response: Video information in a bandwidth of 1 cps to 220 k cps shall be received with a flat response +1, -1.5 db at the unfiltered output of the demodulator. The demodulator post detection filter shall have a bandpass of dc to 220 k cps, and its amplitude response shall be down 3 db ±0.5 db at 220 K cps. The filter overshoot in response to a step change at the input shall be ≤ 1% of final value.

e) FM SNR Improvement: The post detection SNR for a low frequency sinusoidal signal producing 310 kc peak carrier deviation shall be ≥ 38 db PK-PK/RMS, for predetection SNR ≥ 11.5 db RMS/RMS.

f) Demodulator Transient Characteristics: For a step change in frequency input to the demodulator, (10-90)% rise time shall be ≤ 2 μ secs. No more than 3% overshoot shall occur at the unfiltered output.

6.2.1.2 Narrow Band Television

a) Nominal Center Frequency: 10 mc at input to demodulator (translated to 70 kc within CDC)

b) Maximum Carrier Deviation: ± 5 kc

c) Predetection Filter Bandwidth: -2.5 db ±0.5 db bandwidth =10.5 kc; equivalent noise bandwidth 11.8 kc ±1.1 kc.

d) Post Detection Video Response: Video information in a bandwidth of 0.1 cps to 1.2 k cps shall be received with a flat response of ±1 db at the unfiltered output of the demodulator. The demodulator post detection filter shall have a passband of dc to 1.2 k cps, and its amplitude response shall be that of a linear phase function, down 3 db ±0.5 db at 1.2 k cps.

e) FM SNR Improvement: The post detection SNR for a low frequency sinusoidal signal producing 1.5 kc peak carrier deviation shall be ≥ db PK-PK/RMS for carrier predetection SNR ≥ 5.5 db RMS/RMS.

f) Demodulator Transient Characteristics: For a step change in frequency input to the demodulator the rise time (10-90)% of the filtered output shall be ≤ 300 μ secs. No more than 3% overshoot shall occur at the unfiltered output.

6.2.1.1.3 Wideband FM/FM Mode

a) The post detection SNR shall be as noted below, for the six multiplexer accelerometer channels.

PEAK CARRIER DEVIATION			POST DETECTION SNR DB PK-PK/RMS	
1)	± ______	kc	______	db
2)	± ______	kc	______	db
3)	± ______	kc	______	db
4)	± ______	kc	______	db
5)	± ______	kc	______	db
6)	± ______	kc	______	db

for a low frequency sinusoidal signal, at a predetection SNR ≥ 11.5 db RMS/RMS. The post detection SNR is defined at the second FM demodulator output.

b) The total harmonic distortion of a low frequency sinusoidal signal shall be ≤ %.

6.2.1.1.4 FM/TM Mode: Nominal frequency range - 300 cps - 500 kc (within ______ db) allowable analog signal harmonic distortion ≤ percent maximum.

6.2.1.2 Subcarrier Discriminators (Spec. No. 222210, Rev. C): The following discriminators with proper channel selectors shall be furnished:

Center Frequency KC	Channel Selector BW (-2.5 db) (cps)
2.3	346
1.7	256
5.4	422
5.4	810
3.9	580
7.35	1102
33.0	4420
3.9	580
0.56	22.8
0.96	144
52.5	9900
70.0	9900

Center Frequency KC	Channel Selector BW (-2.5 db) (cps)
90.0	9900
110.0	9900
130.0	10920
150.0	9900

6.2.1.3 Decommutator (Spec. No. 222514, Rev. J): The function of the PCM decommutator is to separate commutated telemetry data received from the S/C to provide individual outputs suitable for display, recording and real-time transmission to the DSIF.

a) PCM input design criteria:

1. Serial bit rates of NRZ bit stream shall be be 4400, 1100, 550, 137.5 and 17.2 BPS.

2. Serial word length shall be adjustable from 5-16 bits including those for synchronization and parity.

3. Word frame length shall be adjustable from 5 to 128 words including frame sync words.

4. Odd parity shall be provided for all words.

b) TV-PCM Mode: The decommutator shall operate with an input in which 16 channel Frame ID PCM data will immediately follow a TV video signal.

6.2.1.4 Telemetry Subsystem BER

a) The PCM telemetry subsystem shall have a BER at a predetection SNR of 10 ± 1 db $\leq 3 \times 10^{-3}$, for all bit rates

b) The TV-PCM mode (TV Commutator Frame ID) shall have a BER $\leq 3 \times 10^{-3}$ at a predetection SNR ≥ 11.5 db for all bit rates, for the normal mode, and at a predetection SNR $\geq$ ____ db for the emergency mode.

6.2.2 Command Subsystem (Spec. No. 222520, Rev. G): The command subsystem generated all digital commands that control the S/C operation with five functional modes: keyboard mode, tape reader mode, auxiliary mode and emergency mode.

6.2.2.1 Format

a) Direct Commands - 24 bit word consisting of 4-bit sync, 5-bit address complement, 5-bit address, 5-bit command complement and 5-bit command. Manchester coding is used on all except the first four bits.

b) Quantitative Commands - Same as a) except substituting a special 10-bit quantity message for the 10-bit command and its complement.

c) Fill in Words - Same sync 4 bits as a) following with 20 bits all ones.

6.2.2.2 Rate and Subcarrier Frequency: The subsystem shall be capable of driving an SCO at 48 BPS. The output frequency of the SCO shall be 2127 ±15 cps for "0" bit and 2473 ±15 cps for a "1" bit.

6.2.2.3 Command Generator (Spec. No. 222521, Rev. G)

a) Command programmer oscillator: 384 cps ±.05 percent,

b) Countdown ratios: 4:1 and 8:1 forming two square waves, 96 cps ±.05 percent and 48 cps ±.05 percent.

c) Square wave rise time: 10 μ sec.

c) Keyboard Entry: Keyboard shall contain an array of eight pushbuttons numbered zero through seven.

e) Command Generator output: Serial train of Manchester coded binary bits having 12 volts dc level for a one bit and 0 volts dc level for a zero bit.

6.2.2.4 Command Printer (Spec. No. 222527, Rev. C): The command printer converts the binary-coded command word into a real-time printout of the transmitted command and the time of command transmission.

a) The printed record shall consist of 11 columns of print in a 12-column format.

b) Characters: 0.085 in. wide - 0.100 in. high

c) Printing rate: ≥ 5 lines per second

d) Input signal code: parallel 1248 BDC code format

e) Input signal logic level: 12^{-6}_{+18} volts for a one bit, 0 ±1 volt for a 0 bit.

6.2.2.5 Punched Tape Reader (Spec. No. 222522, Rev. D): The tape reader provides a semi-automatic command word entry mode for the Command System.

a) Tape width: 0.6875 - 1.0 inch width

b) Tape speed: the unit shall be capable of reading at least 30 characters per second in a stepping mode synchronized to an external system.

c) Tape slew speed: 50 characters/sec.

d) Output voltage: 12 ±1 volts code hole; 0±1 volt no code hole.

e) Start time: the unit shall step to the next character and reach a quiescent level in less than 20 ms after receiving a start signal.

6.2.2.6 Subcarrier Oscillator (Spec. No. 222528, Rev. E): The unit shall accept binary-coded command words from the various insertion modes and develop an AF signal (whose output frequency is varied between two adjustable extremes).

a) Output frequency: See 6.2.2.2

b) Deviation: See 6.2.2.2

c) Phasing: increasing positive input voltage shall produce an increasing output frequency

d) Output voltage: adjustable from 0 to 4.5 volts rms.

e) Linearity: when the output frequency is plotted versus .5 volt steps of the input voltage, within the 5 volt operating signal range, deviations from the best straight line shall not exceed ±1 percent.

f) Stability: the change in center frequency shall be less than 1 percent of bandwidth, during an 8 hour period after a 15 minute warm-up.

6.2.2.7 Automatic Tape Entry Mode - Command Subsystem TV

a) Frame Rate: Nominal frame rate for CDC S/C synchronized tape modes shall be one frame every 3.6 sec. in the normal mode. The automatic synchronized command system shall cause a maximum time delay of equivalent to one frame in fifty during a prolonged sequence.

b) Number of command without changing tape: A tape reader-spooler combination shall be capable of storing up to 10,000 commands without changing spools or switching to an alternate configuration.

6.2.2.8 Command Subsystem Uplink BER

a) Keyboard Mode: The command subsystem shall have a BER, at the SCO input terminals 1×10^{-5}.

b) Automatic tape entry mode: The command subsystem shall have a BER at the SCO input terminals $\leq 1 \times 10^{-5}$.

6.2.3 Recording Interfaces: The CDC shall provide suitable buffering to record on playback at specified signal levels on the following DSIF recorders,

CEC 36 Channel Oscillograph
FR 800
FR 1400

CDC signals to be recorded on FR 1400 and FR 800 are summarized in Table and signal characteristics are listed in Table . Telemetry signals to be recorded on the CEC oscillograph are listed in Table .

6.2.4 Television Subsystem (Spec. No. 222530, Rev. C) This system displays and photographically records S/C TV data.

TABLE II-1. SUMMARY OF SIGNALS TO BE RECORDED ON FR-1400 AND/OR FR-800

Signal	Output Impedance	Tape Recorder Used/Impedance	Bandwidth	Signal Format	Bit Rates	Signal Level	Record Direct Or FM
1. Command SCO	20	1000 -FR-1400 10,000 -FR-800	Audio Frequency	Digital	48 modulating	3.77V RMS	direct
2. Phase Detected Telemetry	20	1000 -FR-1400	250 CPS-160 KC (Spec)	NRZ Digital On Sub-Carrier Oscil-lators	17.2 - 4400 mod-ulating SCO's from .56 KC to 33 KC	As shown in Table II Amplified by 1, 10, or 100	direct
3. Reconstructed telemetry	1 100	1000 FR-1400	17 BPS -4400 BPS	NRZ Digital	17.2 - 4400	-1 to -11 volts	square wave FSM
4. Wideband Telemetry A. Accel-erometer channels	20	1000 FR-1400	47.5 KC-160 KC	Analog	-	1.23V RMS	direct
B. Video	20	1000 FR-1400	0-220 KC	Analog	-	0.764V RMS	direct
C. 33 KC SCO (FM)	20	1000 FR-1400	28.5 KC-37.5 KC	NRZ	4400 on 33 KC SCO	0.194V RMS	direct

NOTE

The DSIF TVGDHS - Television Ground Data Handling System - Document S. O. and R. No. FOT-2-110 - is the primary system. The HAC TV subsystem specified here shall be considered backup to the primary system.

6.2.4.1 Normal TV Monitor and Photo Recorder

6.2.4.1.1 Monitor Display: Each CDC shall provide at least one real time display of the TV image. The primary purpose of the display is to allow the console operator to monitor and verify proper operation of the cameras. During surveys of selected areas, manual control may very well be used; hence, the monitor display should be of high enough quality to allow the console operator to perform the focus and (if necessary) iris control functions. However, display of the full resolution inherent in the video signal is not a requirement.

a) Output at video monitor in response to step input pulse from the S/C: ≤ 5 percent overshoot.

b) Peak horizontal sync jitter at predetection SNR of 11.5 RMS/RMS db: ≤±3 pixels.

6.2.4.2 Emergency TV Monitor and Photo Recorder

a) Overshoot in response to a step change at S/C system input shall be ≤ 5%.

b) Peak horizontal sync jitter at predetection SNR of 5.5 db: ≤ ± 5 pixels.

6.2.4.2.1 Photographic Recording: The entire photographic recording system is required to achieve a relative response (square wave) as follows:

N(TV lines)	Relative Response
200	0.9
400	0.45
500	0.3
600	0.2

The recording system must be capable of photographically sorting both the video signal and the frame identification data, excluding vidicon and electronic temperature information.

6.2.5 System Tester: The system tester simulates the S/C for the purpose of testing the CDC. Specifications or major system tester components are:

- 274004 - S-Band Transponder
- 274006 - Command Decoder
- 223160 Rev. E - Random Noise Generator
- 223154 Rev. C - TV/PCM Generator
- 223155 Rev. C - IF Oscillator Test Set
- 288723 Rev.A - Command and Data Handling System Test Set

6.3 Orbit Determination: Determination of the transit trajectory shall be accomplished at the SFOF based upon doppler velocity and angle position data.

6.3.1 Accuracy Requirement at Impact: The three sigma error of the last orbit determination done prior to the sending of terminal maneuver commands shall result in no greater than the following errors in the unbraked impact variables.

6.3.1.1 Time of Impact: ±10 sec [(2)] (required); ±3 sec [(2)] (desired).

6.3.1.2 Magnitude of Unbraked Impact Velocity: ±3 meters/sec [(2)].

6.3.1.3 Direction of Unbraked Impact Velocity: < 0.18 deg(1) for vertical impact trajectories; < 0.40 deg(1) for 45 degree impact trajectories (this corresponds to a 30 km(1) miss vector error).

6.3.2 Timing Requirements on Orbit Determinations

6.3.2.1 The first orbit determination shall be completed no later than 4 hours after launch.

6.3.2.2 The orbit determination to be used for the midcourse correction computation shall be completed at least 2 hours prior to the time the midcourse correction is to be implemented.

6.3.2.3 The orbit determination to be used for the terminal descent calculations shall be completed at least 3 hours prior to unbraked impact.

6.3.3 Standard Trajectory Data: Post injection standard trajectory data shall be supplied by the S/C contractor to JPL at least seven weeks prior to launch. There shall be a minimum of three trajectories across the permissible launch window for each launch day.

6.4 Midcourse Guidance Program: The midcourse correction computations shall be performed at the SFOF. Programs for the normal midcourse correction as well as those for emergency procedures under abnormal miss conditions or certain abnormal S/C conditions shall be supplied by the S/C contractor.

6.4.1 Computation Technique: The midcourse correction computation technique shall be an iterative procedure in which storage of influence coefficients is not required.

6.4.2 Inputs: The inputs to the midcourse guidance program shall be the best estimate injection conditions, the desired landing site, time at which the midcourse correction is to be made, as well as other pertinent S/C and system parameters. These constant parameters shall be supplied no later than 2 months prior to launch.

6.4.3 Outputs: The outputs of the midcourse guidance program shall be the two maneuver angles, the velocity increment, the time tolerance over which the correction is to be made, emergency fuel requirements and emergency landing site locations.

6.4.4 Computation Time: This program shall require no more than 15 minutes running time on the SFOF computer.

6.4.5 Midcourse Guidance Computation Error: The contribution to the midcourse correction accuracy (Section 3.6.3) due to the midcourse guidance computation program shall be less than 5 km.

6.5 Terminal Guidance Program: The terminal guidance computations shall be performed at the SFOF. Programs for the normal terminal guidance computations as well as those for emergency procedures shall be supplied by the S/C contractor.

6.5.1 Computation Technique: The terminal guidance computations shall be performed with an iterative procedure in conjunction with the best model available which describes the terminal phase.

6.5.2 Inputs: The inputs to the terminal guidance program shall be the best estimate trajectory data and other pertinent S/C and system parameters. These constant parameters shall be supplied no later than 2 months prior to launch.

6.5.3 Outputs: The outputs of the terminal guidance program shall be the three maneuver angles for the S/C, the time for marker radar actuation, the engine ignition time delay, the vernier engine thrust level, and the planar array polar angle.

6.5.4 Computation Time: This program shall require no more than 10 minutes running time on the SFOF computer.

6.6 Engineering Data Reduction: It is required that S/C telemetry data be presented to S/C performance specialists at the SFOF in a manner readily interpretable for determining S/C status and planning sequence modifications under non-standard conditions. Data processing requirements include detection of bad parity, scaling to engineering units, reference voltage and functional calibration, alarm monitoring; and accuracy estimation. Real time (~ 1 second delay) alarm monitoring and limited processing are required for some channels during critical sequences. SFOF computer programs shall be developed to satisfy these requirements.

6.6.1 Inputs: Input data consists primarily of raw S/C telemetry after demodulation and decommutation.

6.7 Power Management Program: Estimates of present battery charge conditions and predictions of future battery charge conditions are required to assist sequence planning during transit and lunar phases. A SFOF computer program shall be developed for this purpose.

6.7.1 Inputs: The program shall operate from the following input data:

1) Commands and times already sent to S/C
2) Operational events and times in the planned sequence
3) Telemetered S/C power system data
4) Temperatures of power system components from Thermal Management Program.

6.7.2 Outputs: The program shall provide the following data as principal output:

1) Estimated battery charge condition

2) Variance in estimated battery charge condition

3) Power dissipation versus time in specified components for use by the Thermal Management Program.

6.8 Thermal Management Program: Predictions of various S/C component temperatures are required to assist sequence planning in non-standard conditions. A SFOF computer program shall be developed for this purpose.

6.8.1 Inputs: The program shall operate from the following input data:

1) Telemetered S/C temperatures

2) Power dissipations from Power Management Program

3) Shadowing of components

4) Component view factors to space

5) Sun angles and projected areas of components.

6.8.2 Outputs: The program shall provide predicted component temperatures.

6.9 Attitude Program: Many aspects of S/C performance and scientific data interpretations are influenced by the S/C attitude with respect to the lunar surface, the sun, and the earth. A SFOF computer program shall be developed to determine S/C attitude after landing and provide various attitude related data.

6.9.1 Inputs: The program shall operate from the following principal input data:

1) Time at which computation is to be performed

2) Ephemeris of sun and earth in lunar coordinates

3) Nominal roll attitude at touchdown

4) Latitude and longitude of landing site

5) Telemetered solar panel and planar array gimbal angles and estimated dispersions in these angles

6) Maximum lunar surface slope expected

7) Star directions in TV camera coordinates

8) Gimbal angles immovable or otherwise specified.

6.9.2 Outputs: The program shall provide the following output data:

1) Nominal sun direction after landing and corresponding gimbal angle.

2) Loci of possible earth directions and corresponding gimbal angles after sun direction has been determined.

3) Attitude matrix describing the orientation of the S/C in selenographic coordinates.

4) Angle between S/C roll axis and local vertical and error estimate.

5) Gimbal angles for repositioning of solar panel and planar array.

6) Optimum gimbal angles and deviations for non-standard planar array and solar panel orientations and drive failures.

6.10 Telecommunications Program: Predictions of signal-to-noise ratio margins in the DSS receiver as a function of trajectory and component parameters for all Surveyor data and command modes are required to assist sequence planning in non-standard conditions. A SFOF computer program shall be developed to provide this data.

6.10.1 Inputs: The program shall operate from specified S/C and DSS parameters and trajectory data.

7.0 TRAJECTORY AND GUIDANCE REQUIREMENTS

7.1 Boost and Injection Phase

7.1.1 The boost and injection trajectory is that portion of the trajectory starting at lift-off and ending with injection into the trans-lunar trajectory.

7.1.2 Launch Site: The launch site shall be complex 36 of the Cape Kennedy, Florida facilities of the Air Force Eastern Test Range (AFETR).

7.1.2.1 Pad 36A

Geocentric	Latitude	28.3106 deg. North
	Longitude	279.4618 deg. East

7.1.2.2 Pad 36B

Geocentric	Latitude	28.3075 deg. North
	Longitude	279.4588 deg. East

7.1.4 Landing Area of Interest: For the purpose of selecting launch days on which trajectories will be designed, the area of interest for Surveyor landings is in the region bounded on the west by 45° selenographic longitude and on the east by the off-vertical landing capability of the spacecraft.

7.1.5 Launch Day Restrictions: For the purpose of selecting launch days on which trajectories will be designed those days which meet the following criteria shall be considered:

7.1.5.1 Direct Ascent Missions

7.1.5.1.1 Launch window greater than zero minutes

7.1.5.1.2 Launch vehicle flight performanc reserve greater than zero pounds.

7.1.5.1.3 Launch azimuth between 80 and 115 degrees east of true north.

7.1.5.1.4 Landing between 20 hours before the morning terminator and 72 hours before the evening terminator in the area of interest defined in Paragraph 7.1.4.

7.1.5.2 Parking Orbit Missions

7.1.5.2.1 Launch window greater than zero minutes

7.1.5.2.2 Centaur parking orbit coast time between 116 seconds and 25 minutes.

7.1.5.2.3 Launch azimuth between 78 and 115 degrees east of true north.

7.1.5.2.4 Landing between 20 hours after the morning terminator and 150 hours before the evening terminator in the area of interest defined in Paragraph 7.1.4.

7.1.6 Launch Azimuth Versus Time of Lift-Off: On each day for which launch vehicle guidance constants are to be determined, the implementation of the exact relationship between launch azimuth and the time of lift-off shall be the responsibility of the launch vehicle contractor.

7.1.7 Post Lift-Off Trajectory Parameters: For nominal 66 hour trajectories, the parameters shall be as follows:

7.1.7.1 For direct ascent trajectories:

Twice Vis Viva Energy (C_3): -0.85 to -1.70 Km^2/sec^2

Nominal Perigee Altitude: 90 n. mi (167 Km)

Injection True Anomaly: -8 to +15 degrees

Nominal Boost Arc: 27.5 degrees

7.1.7.2 <u>For parking orbit trajectories</u>:

Twice Vis Viva Energy (C_3): -0.85 to -1.85 Km^2/sec^2

Nominal Parking Orbit Altitude: 90 n. mi (167 Km)

Parking Orbit Coast Time: 126 seconds to 25 minutes

Nominal First Boost Arc: 20 degrees

Nominal Second Boost Arc: 8 degrees

7.1.8 <u>Specification of Target Criteria</u>:

7.1.8.1 <u>Target Criteria</u>

Launch Days - calendar day of launch

Impact Speed - unbraked lunar impact speed

Impact Loaction - selemographic latitude and longitude of unbraked impact

Arrival Time Constraints - earliest and latest time of unbraked impact

7.1.8.2 <u>Targeting Procedure:</u> The trajectories shall be targeted for each day so that unbraked lunar impact shall occur at the specified speed and location. If, however, for any trajectory the specified speed cannot be obtained without violating an arrival time constraint, that trajectory will be targeted so that unbraked impact will occur at the specified location and the arrival time constraint.

7.1.8.3 Trageting Accuracy: No trajectory shall require more than a 2 meter per second midcourse maneuver at 20 hours after injection to achieve the specified unbraked impact location and no more than a 4 meter per second midcourse maneuver at 20 hours after injection to achieve all the specified target criteria.

7.2 Transit Phase

7.2.1 Transit Time:

7.2.1.1 Direct Ascent Trajectories: For a nominal [illegible] hour trajectory, the actual nominal time of flight from injection to unbraked impact varies from [illegible] to [illegible] hours.

7.2.1.2 Parking Orbit Trajectories: For a nominal [illegible] hour trajectory, the actual nominal time of flight from injection to unbraked impact varies from [illegible] to [illegible] hours.

7.2.2 Goldstone Visibility at Landing: All transit trajectories shall be designed so that the spacecraft will be visible to the Goldstone Pioneer Station (DSIF 11) a minimum of 2 hours prior to and 3 hours following unbraked impact. Visibility is defined by the mutual satisfaction of the following constraints: an elevation angle of 5 degrees, antenna gimbal angle constraints and terrain interference masks.

7.2.3 Unbraked Impact Speed Limits: The targeted unbraked lunar impact speed shall be between 2615 and 2692 meters/second for direct ascent trajectories and 2600 and 2690 meters/second for parking orbit trajectories.

8.0 SERVICE LIFE AND RELIABILITY OBJECTIVES

8.1 The service life of the spacecraft as defined in paragraph 3.14.1 of JPL Specification 30240 shall be as an objective, 90 Earth days.

8.2 The reliability requirements of the spacecraft are defined in paragraph 3.14.2 of JPL Specification 30240.

8.3 The detailed subsystem and component reliability objectives and concomitant program plan which will insure the achievement of the Mission reliability objectives are presented in HAC Specifications 224600, 224601, and 224602.

~~ENGINEERING ORDER~~ / REVISION NOTICE	HUGHES AIRCRAFT CO. CULVER CITY, CALIF. CODE IDENT 82577	224510 [redacted] SPEC NUMBER
DWG / SPEC TITLE FUNCTIONAL REQUIREMENTS	☐ EO NO. ____ SH ____ OF ____ ☒ REV NOTICE LTR H SH ____ OF ____	
CONTROL ITEM NAME Software	AUTHORITY * RECORD	
PRODUCT DESIGNATION Surveyor	CHANGE EFFECTIVITY (LOT NO.)	

CHANGE EFFECTIVITY (CONTROL ITEM SERIAL NO.)

CONTROL ITEM PART NUMBER	SERIAL NUMBER	CONTROL ITEM PART NUMBER	SERIAL NUMBER	CONTROL ITEM PART NUMBER	SERIAL NUMBER

SUFFIX NO.	DESCRIPTION

* Incorporate the following changes into an H revision of the Functional Requirements:

EO ECA
31684 - 113057
32202 - 112831
32211 - 112961-31
32213 - 112836-7
32216 - 113249
32217 - 112809
32218 - 113248
32219 - 113078
51488 - 112836-4
51491 - 112943
51492 - 112890
51496 - 112795-4
51499 - record change
51577 - 113179
51580 - 113062
Δ 32222 - 113321-1

NOTE: A "G" Revision of the 224510 was released on 2-9-67 by incorporating 224510 F and the "F" outstanding EO's (all the EO's listed above with the exception of the EO having a Δ along side of it.) After release of 224510G numerous typing errors were found in the "G" text and Engineering Stop Order 32224 (2-16-67) was used to nulify and recall the entire "G" revision distribution. On 3-3-67 EO 32222 was released against the "F" specification. The "H" revision is now created, with this R/N, by incorporating all of the EO's listed above against the "F" revision of the specification.

L/M: LIST OF MATERIALS F/D: FIELD OF DRAWING G/N: GENERAL NOTES

DISPOSITION OF ITEMS

[signature]	3/3/67	[signature]	3/3/67		
PREPARED BY	DATE	ENGRG APPROVAL	DATE		DATE
		N. J. Capon	3-3-67		
CHECKED BY	DATE	RELEASE GROUP	DATE		DATE

UNCLASSIFIED

ENGINEERING ORDER / REVISION NOTICE	HUGHES AIRCRAFT CO. CULVER CITY, CALIF. CODE IDENT 82577	224510 DWG / SPEC NUMBER
DWG / SPEC TITLE DETAIL SPECIFICATION SURVEYOR SYSTEM FUNCTIONAL REQUIREMENTS	☒ EO NO. 32232 SH 1 OF 1 ☐ REV NOTICE LTR ___ SH ___ OF ___	
CONTROL ITEM NAME MAIN RETRO ENGINE	AUTHORITY ECR 3391R4 ECN 113387-1	
PRODUCT DESIGNATION SURVEYOR	CHANGE EFFECTIVITY (LOT NO.) SC-3 ONLY (A22-8)	

CHANGE EFFECTIVITY (CONTROL ITEM SERIAL NO.)

CONTROL ITEM PART NUMBER	SERIAL NUMBER	CONTROL ITEM PART NUMBER	SERIAL NUMBER	CONTROL ITEM PART NUMBER	SERIAL NUMBER
238612-1	A22-8 (SC-3)	[redacted]			
263723	1				

SUFFIX NO.	DESCRIPTION
1.	PAR. 4.2.1.3 WAS: THE MAXIMUM INSTANTANEOUS THRUST OVER - - - SHALL NOT EXCEED 10,000 POUNDS. IS: THE MAXIMUM INSTANTANEOUS THRUST OVER - - - SHALL NOT EXCEED 10,300 POUNDS. VARIANCE, DO NOT INCORPORATE.

L/M: LIST OF MATERIALS F/D: FIELD OF DRAWING G/N: GENERAL NOTES

DISPOSITION OF ITEMS USE

O. Murray	2-16-67	[illegible]	4/17/67	C J Cole	3/16/67
PREPARED BY	DATE	ENGRG APPROVAL	DATE		DATE
[illegible]	2-16-67	[illegible]	[illegible]		
CHECKED BY	DATE	RELEASE GROUP	DATE		DATE

ENGINEERING ORDER / REVISION NOTICE	HUGHES AIRCRAFT CO. CULVER CITY, CALIF. CODE IDENT 82577	224510 H DWG / SPEC NUMBER
DWG / SPEC TITLE Functional Requirements	EO NO. 32223 SH. 1 OF 6; REV NOTICE LTR ___ SH. ___ OF ___	
CONTROL ITEM NAME Software	AUTHORITY ECR 354840/ECA 1344-1, CL I	
PRODUCT DESIGNATION Surveyor	CHANGE EFFECTIVITY (LOT NO.) SC-3 and up	

CHANGE EFFECTIVITY (CONTROL ITEM SERIAL NO.)

CONTROL ITEM PART NUMBER	SERIAL NUMBER	CONTROL ITEM PART NUMBER	SERIAL NUMBER	CONTROL ITEM PART NUMBER	SERIAL NUMBER
224312-4	13 & UP				

SUFFIX NO.	DESCRIPTION
1	Revise paragraph 5.2.2.9 as follows:

WAS:

5.2.2.9 Normal Mode System Signal-to-Noise Ratio

The peak to peak signal to rms noise ratio, as defined below, shall be 27 db or greater, including vidicon ...

IS:

5.2.2.9 Normal Mode System Signal-to-Noise Ratio

The peak to peak signal to rms noise ratio, as defined below shall be 31 db or greater, including vidicon ...

2 Revise paragraph 5.2.3.6 as follows:

WAS:

5.2.3.6 Normal Mode Camera Signal-to-Noise Ratio

The peak-to-peak signal to rms noise ratio shall be 28 db or greater, as defined in 5.2.2.9, and shall be consistent with the horizontal relative response as shown in Figure 5-5. No periodic electronic noise shall have peak-to-peak amplitudes greater than 8 per cent V_s (see Figure 5-1).

IS:

5.2.3.6 Normal Mode Camera Signal-to-Noise Ratio

The peak-to-peak signal to rms noise ratio shall be 32 db or greater, as defined in 5.2.2.0, and shall be consistent with the horizontal relative response as shown in Figure 5-5. No periodic electronic noise shall have peak-to-peak amplitudes greater than 8 per cent V_s (see Figure 5-1).

L/M: LIST OF MATERIALS F/D: FIELD OF DRAWING G/N: GENERAL NOTES

DISPOSITION OF ITEMS

PREPARED BY	DATE	ENGINEERING	DATE		
[illegible]	[illegible]	[illegible]	[illegible]		
CHECKED BY	DATE	MANAGEMENT	DATE		

FORM

ENGINEERING ORDER / REVISION NOTICE CONTINUATION SHEET	HUGHES AIRCRAFT CO. CULVER CITY, CALIF. CODE IDENT 98677	224510 DWG / SPEC NUMBER
DWG / SPEC TITLE: Functional Requirements	☐ EO NO. 32223 SH 2 OF 6	☐ REV NOTICE LTR ___ SH ___ OF ___

SUFFIX NO.	DESCRIPTION
3	Change Figure 5-2 as indicated by pages 3 and 4 of this EO.
4	Change Figure 5-5 as indicated by pages 5 and 6 of this EO.

L/M LIST OF MATERIALS   F/M FIELD OF DRAWING   G/M GENERAL NOTES

Was:

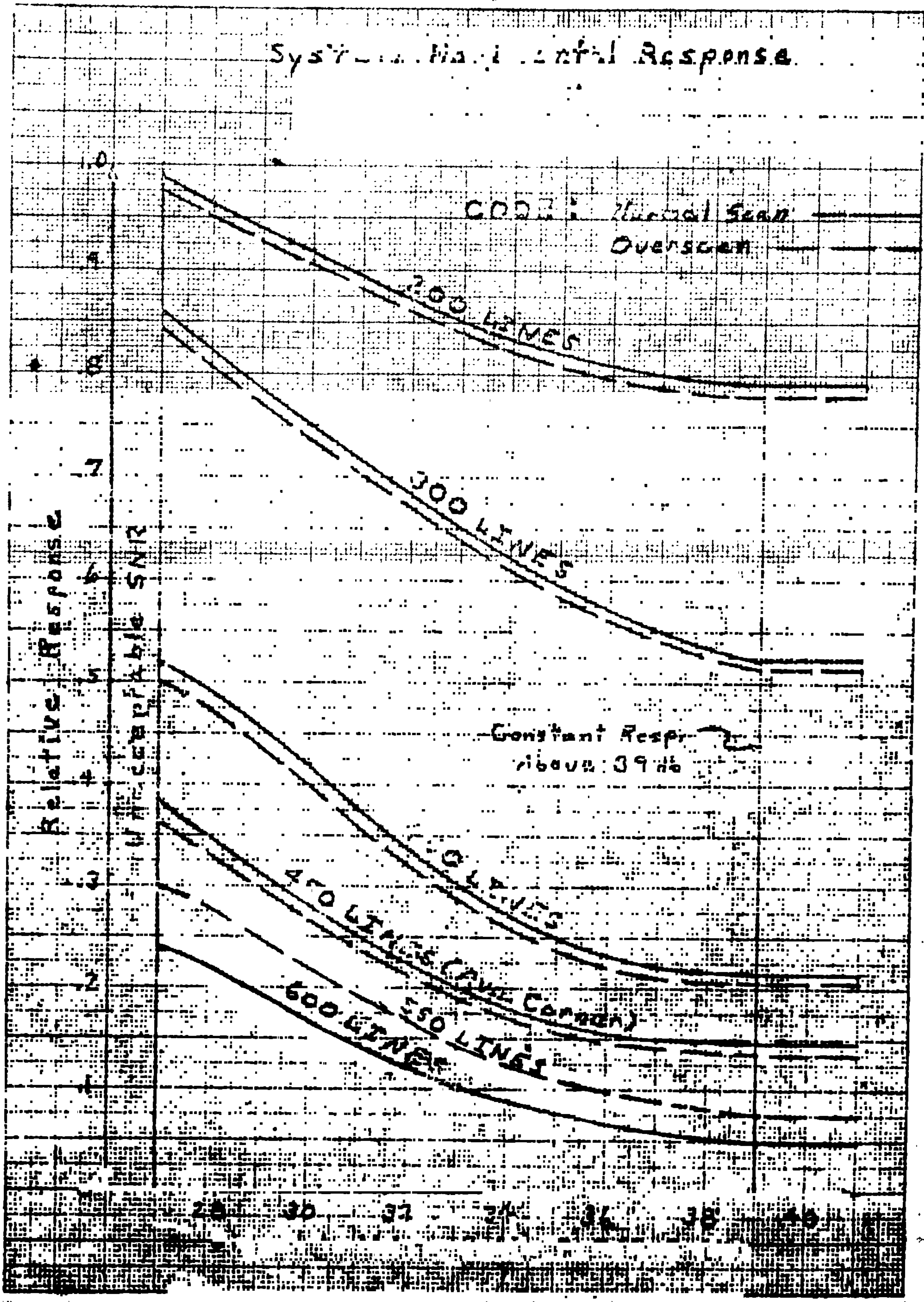


Figure 5.2

Is:

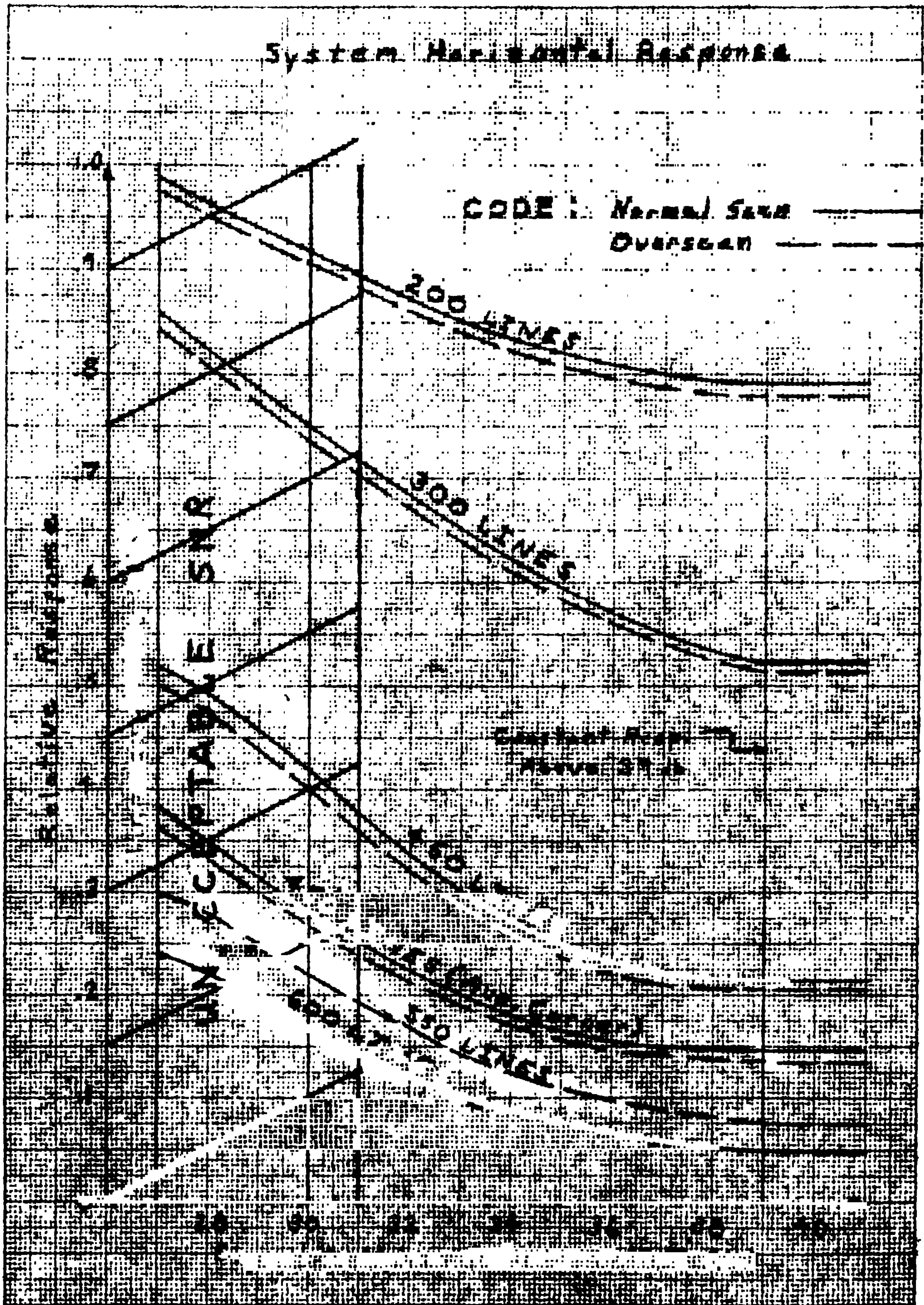

System Horizontal Response
CODE: Normal Scan
Overscan
200 LINES
300 LINES
UNACCEPTABLE SNR
Relative Response
1.0
.9
.8
.7
.6
.5
.4
.3
.2
.1
550 LINES
600 LINES

Figure 5.2

Ves:

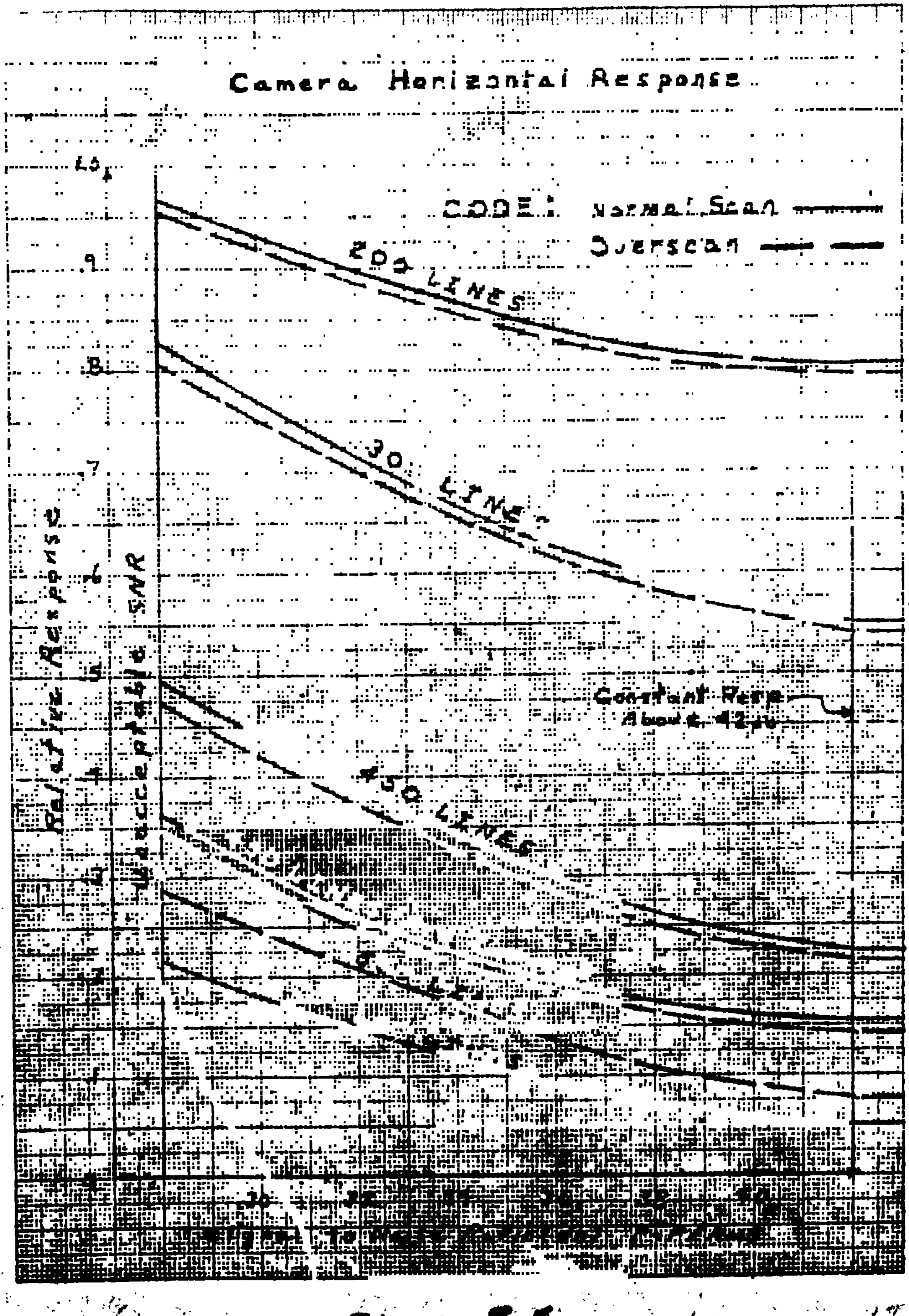

Camera Horizontal Response
CODE: Normal Scan
Overscan
200 LINES
30 LINES
450 LINES
Relative Response
Unacceptable SNR
.9
.8
.7
.6
.5

Figure [illegible].5

Is:

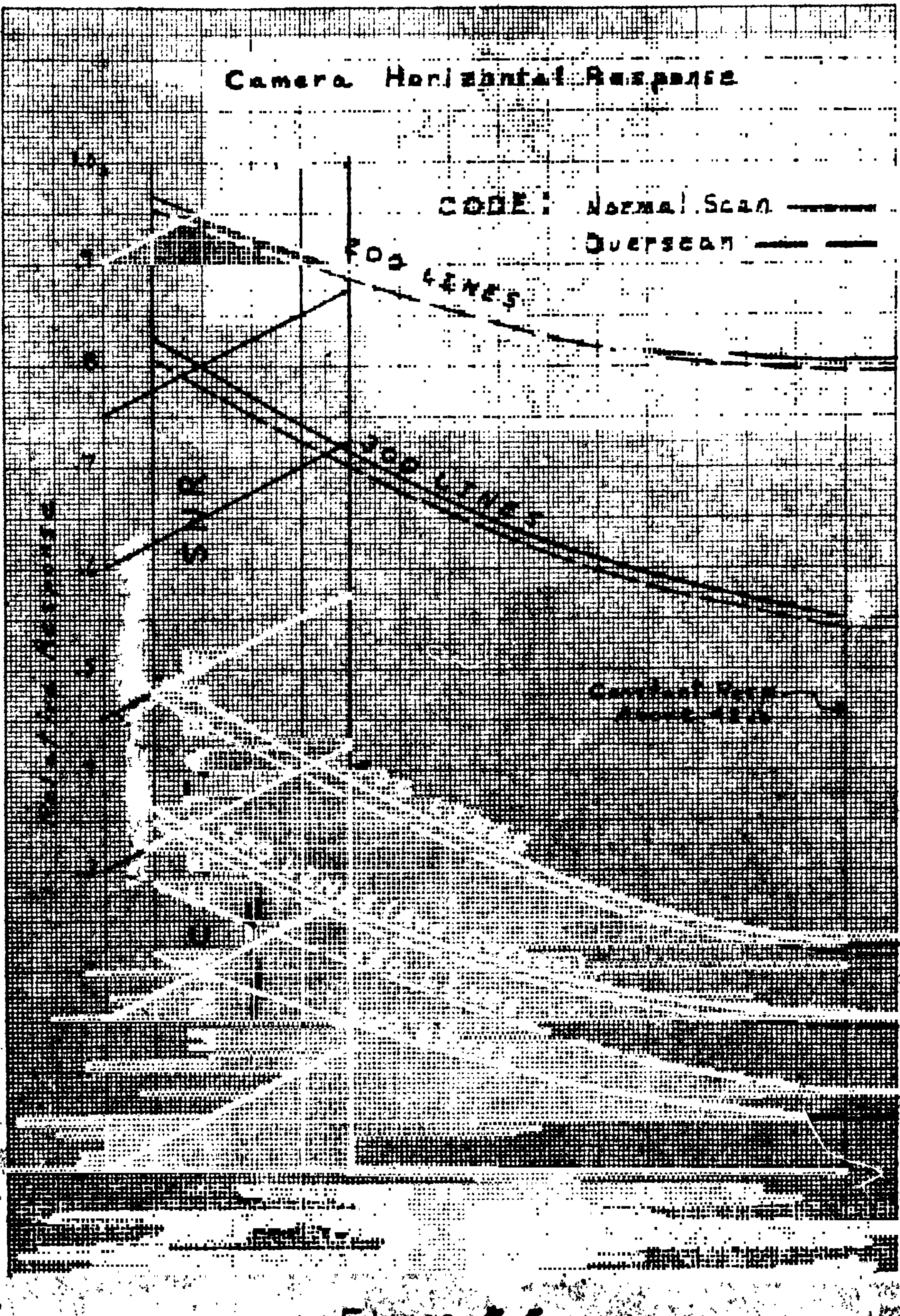


Figure 5.5